-CR 137505

[redacted] CR [redacted]
TRW Document No. 2291-6006-RU-00

Systems Design Study of the Pioneer Venus Spacecraft

Final Study Report

Volume I. Technical Analyses and Tradeoffs Sections 5-6 (Part 2 of 4)

(NASA-CR-137505) SYSTEMS DESIGN STUDY OF THE PIONEER VENUS SPACECRAFT. VOLUME 1. TECHNICAL ANALYSES AND TRADEOFFS, SECTIONS 5-6 (PART 2 OF 4) Final Study (TRW Systems Group) 229 p HC $14.50 CSCL 22B G3/31
N74-32305
Unclas
47082

29 July 1973

Contract No. NAS2-7249

Prepared for

AMES RESEARCH CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

TRW
SYSTEMS GROUP

MARTIN MARIETTA

Systems Design Study of the Pioneer Venus Spacecraft
Final Study Report
Volume I. Part 2 Sections 5-6
TRW SYSTEMS GROUP

PIONEER VENUS STUDY BOOKMARK

SUBJECT LOCATOR FOR VOLUME 1. TECHNICAL ANALYSES AND TRADEOFFS

SUBJECT	VOLUME 1 LOCATION SECTION	VOLUME 1 LOCATION (PART)	APPENDICES LOCATION (PART)
INTRODUCTION	1	1 OF 4	
SUMMARY	2	1 OF 4	
SCIENCE ANALYSIS AND EVALUATION	3	1 OF 4	1 OF 3
MISSION ANALYSIS AND DESIGN	4	1 OF 4	
SYSTEM CONFIGURATION CONCEPTS AND TRADEOFFS	5	2 OF 4	
SPACECRAFT SYSTEM DEFINITION	6	2 OF 4	1 OF 3
PROBE SUBSYSTEM DEFINITION	7	3 OF 4	2 OF 3
PROBE BUS AND ORBITER SUBSYSTEM DEFINITION AND TRADEOFFS	8	4 OF 4	3 OF 3
NASA/ESRO ORBITER INTERFACE	9	4 OF 4	3 OF 3
MISSION OPERATIONS AND FLIGHT SUPPORT	10	4 OF 4	3 OF 3
LAUNCH VEHICLE-RELATED COST REDUCTIONS	11	4 OF 4	3 OF 3
LONG LEAD ITEMS AND CRITICAL AREAS	12	4 OF 4	

PIONEER VENUS STUDY BOOKMARK

CONFIGURATION SYMBOLS

A/C III	PROBE BUS
A/C IV	
T/D III	
A/C III	ORBITER FIXED DISH ANTENNA
A/C IV	
T/D III	
A/C III	ORBITER DESPUN REFLECTOR FRANKLIN ARRAY
A/C IV	
T/D III	
12 W A/C III	ORBITER FANSCAN FRANKLIN ARRAY-12ω TRANSMITTER POWER
12 W T/D III	
31 W A/C III	ORBITER FANSCAN FRANKLIN ARRAY-31ω TRANSMITTER POWER
31 W T/D III	
A/C III	LARGE PROBE
A/C IV	
T/D III	
A/C III	SMALL PROBE
A/C IV	
T/D III	

LAUNCH VEHICLES

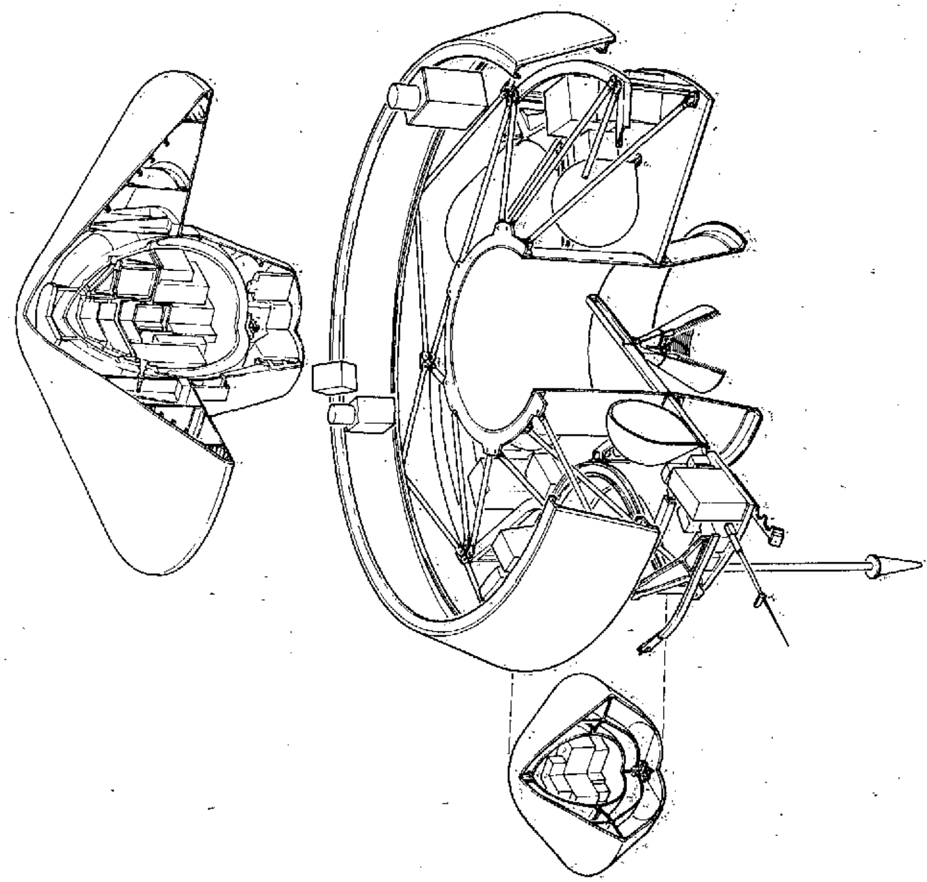
A/C	–	ATLAS/CENTAUR
T/D	–	THOR/DELTA

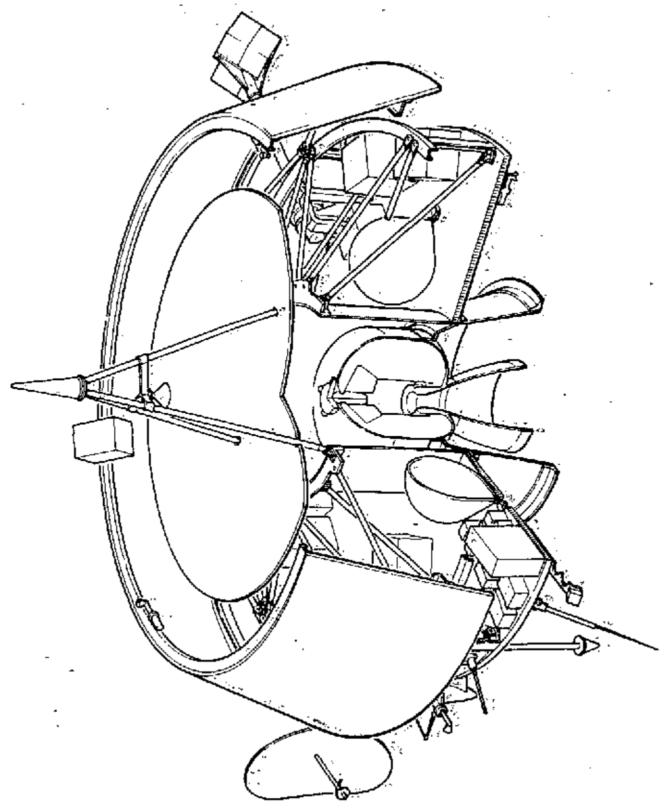
SCIENCE VERSION

III	–	DEFINED IN NASA/AMES LETTER 2 NOVEMBER 1972
IV	–	DEFINED IN NASA/AMES LETTER 13 APRIL 1973

TRW SYSTEMS GROUP

MARTIN MARIETTA





LIST OF VOLUMES

VOLUME I. TECHNICAL ANALYSES AND TRADEOFFS

SECTIONS 1-4 (PART 1 OF 4)

1. Introduction
2. Summary
3. Science Analysis and Evaluation
4. Mission Analysis and Design

VOLUME I. TECHNICAL ANALYSES AND TRADEOFFS

SECTIONS 5-6 (PART 2 OF 4)

5. System Configuration Concepts and Tradeoffs
6. Spacecraft System Definition

VOLUME I. TECHNICAL ANALYSES AND TRADEOFFS

SECTION 7 (PART 3 OF 4)

7. Probe Subsystem Definition

VOLUME I. TECHNICAL ANALYSES AND TRADEOFFS

SECTIONS 8-12 (PART 4 OF 4)

8. Probe Bus and Orbiter Subsystem Definition and Tradeoffs
9. NASA/ESRO Orbiter Interface
10. Mission Operations and Flight Support
11. Launch Vehicle-Related Cost Reductions
12. Long Lead Items and Critical Areas

VOLUME I APPENDICES

SECTIONS 3-6 (PART 1 OF 3)

VOLUME I APPENDICES

SECTION 7 (PART 2 OF 3)

VOLUME I APPENDICES

SECTIONS 8-11 (PART 3 OF 3)

VOLUME II. PRELIMINARY PROGRAM DEVELOPMENT PLAN

VOLUME III. SPECIFICATIONS

CR 137505

[redacted]
TRW Document No. 2291-6006-RU-00

Systems Design Study of the Pioneer Venus Spacecraft

Final Study Report

Volume I. Technical Analyses and Tradeoffs Sections 5-6 (Part 2 of 4)

29 July 1973

Contract No. NAS2-7249

Prepared for

AMES RESEARCH CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

TRW SYSTEMS GROUP

MARTIN MARIETTA

CONTENTS

	Page
5. SYSTEM CONFIGURATION CONCEPTS AND TRADEOFFS	5-1
5.1 Probe Configurations and Tradeoffs	5-3
5.1.1 Preferred Atlas/Centaur Probe Configurations	5-7
5.1.2 Configuration Options	5-9
5.1.2.1 Entry Configuration	5-9
5.1.2.2 Descent Capsule	5-10
5.1.2.3 Staging and Decelerator Options	5-11
5.1.3 Thor/Delta Tradeoffs	5-13
5.1.3.1 Results of Thor/Delta Configuration Studies	5-15
5.1.3.2 Recommended Thor/Delta Prove Configuration Summary	5-17
5.1.4 Atlas/Centaur Tradeoffs	5-17
5.1.4.1 Candidate Configurations	5-19
5.1.4.2 Results of Configuration Studies	5-19
5.1.4.3 Recommended Pre-Midterm Atlas/Centaur Probe Configuration	5-22
5.1.4.4 Development of the Final Atlas/Centaur Preferred Configuration - Evolution from Midterm Design	5-24
5.1.5 Probe Environments	5-25
5.1.5.1 Atlas/Centaur Environments	5-26
5.1.5.2 Thor/Delta Environments	5-26
5.2 Probe Bus and Orbiter Configuration Tradeoffs	5-29
5.2.1 Configurations Designed for Version IV Science and all Atlas/Centaur Launch for Both Missions in 1978	5-29
5.2.2 Configurations Designed Under the Original Study Ground Rules	5-30
5.2.3 Design Drivers	5-32
5.2.3.1 Spin-Axis Orientation	5-32
5.2.3.2 Antenna Selection	5-35
5.2.3.3 Solar Array	5-36
5.2.3.4 Sequential Versus Simultaneous Small Probe Release	5-39
5.2.4 Probe Bus Configuration Details	5-39
5.2.4.1 Thor/Delta Probe Bus Configuration	5-40
5.2.4.2 Atlas/Centaur Preferred Probe Bus Configuration	5-40
5.2.4.3 Alternative Design Based on DSCS-II Spacecraft	5-46

CONTENTS (CONTINUED)

			Page
	5.2.5	Orbiter Configuration Details	5-47
		5.2.5.1 Thor/Delta	5-48
		5.2.5.2 Atlas/Centaur Configurations	5-55
		5.2.5.3 Preferred Atlas/Centaur Orbiter Configuration	5-59
	5.2.6	ESRO Configurations	5-60
		5.2.6.1 Liquid Versus Solid Orbit Insertion Propulsion	5-62
	5.2.7	Spacecraft/Launch Vehicle Mechanical Interfaces	5-63
	5.2.8	Preliminary Design Loads and Environments	5-66
6.		SPACECRAFT SYSTEM DEFINITION	6-1
6.1		Probe System Definition	6-1
	6.1.1	Atlas/Centaur Probes	6-1
		6.1.1.1 Aerodynamic and Flight Dynamic Performance	6-1
		6.1.1.2 Mechanical Design Concept	6-4
		6.1.1.3 Mass Properties	6-11
		6.1.1.4 Electrical Design Concept	6-15
		6.1.1.5 Preferred Probe System Operation	6-21
	6.1.2	Thor/Delta Probes	6-25
		6.1.2.1 Aerodynamic and Flight Dynamic Performance	6-25
		6.1.2.2 Mechanical Design Concept	6-27
		6.1.2.3 Mass Properties	6-28
		6.1.2.4 Electrical Design Concept	6-32
	6.1.3	Mass Properties Management	6-39
6.2		Probe Bus and Orbiter System Definition	6-45
	6.2.1	Atlas/Centaur Configuration (Version IV Science Payload)	6-45
		6.2.1.1 Mechanical Design Concept	6-45
		6.2.1.2 Dynamics and Attitude Control	6-48
		6.2.1.3 Mass Properties	6-50
		6.2.1.4 Electrical Design Concept (Atlas/Centaur)	6-59
	6.2.2	Thor/Delta Configurations	6-76
		6.2.2.1 Mechanical Design Concept	6-76
		6.2.2.2 Dynamics and Attitude Control	6-76
		6.2.2.3 Mass Properties	6-77
		6.2.2.4 Electrical Design Concept	6-85

CONTENTS (CONTINUED)

			Page
6.3	Mission Reliability		6-93
	6.3.1	Probe/Bus Mission Reliability	6-93
	6.3.2	Probe Reliability	6-94
		6.3.2.1 Probe Reliability, Thor/Delta	6-100
		6.3.2.2 Probe Reliability, Atlas/Centaur	6-101
	6.3.3	Probe Bus Reliability	6-102
		6.3.3.1 Tradeoff Studies	6-102
		6.3.3.2 Thor/Delta Configuration	6-103
		6.3.3.3 Atlas/Centaur Configuration	6-106
	6.3.4	Orbiter Reliability	6-108
		6.3.4.1 Thor/Delta Configuration	6-108
		6.3.4.2 Atlas/Centaur Configuration	6-110

ILLUSTRATIONS

		Page
5-1	Probe Configuration and Tradeoff Conclusions	5-2
5-2	Probe Bus and Orbiter Configurations and Tradeoffs Conclusions	5-4
5-3	Final Preferred Probe Configuration	5-7
5-4A	Atlas/Centaur Final Preferred Large Probe Block Diagram	5-8
5-4B	Atlas/Centaur Final Preferred Small Probe Block Diagram	5-8
5-5	Thor/Delta Large Probe Configurations	5-14
5-6	Thor/Delta Small Probe Configurations	5-15
5-7	Thor/Delta Large Probe	5-17
5-8	Thor/Delta Small Probe	5-17
5-9	Recommended Thor/Delta Large Probe Block Diagram	5-18
5-10	Recommended Thor/Delta Small Probe Block Diagram	5-18
5-11	Recommended (Midterm) Atlas/Centaur Large Probe Configuration	5-22
5-12	Recommended (Midterm) Atlas/Centaur Small Probe Configuration	5-22
5-13	Recommended (Pre-Midterm) Atlas/Centaur Large Probe Block Diagram	5-23
5-14	Recommended (Pre-Midterm) Atlas/Centaur Small Probe Block Diagram	5-23
5-15	Evolution of Preferred Atlas Centaur Design	5-27
5-16	Pioneer Venus Missions Preferred Configurations	5-29
5-17	Optional Orbiter and Alternative Bus and Orbiter Configurations for Version IV Science and Atlas/Centaur Launch of Both Missions in 1978	5-30
5-18	Study Configurations Developed Prior to Redirection for Version IV Science and 1978 Launches on Atlas/Centaur. These Low-Cost, Low-Data-Rate Configurations are Viable with Pre-Version IV Science	5-31
5-19	Effect of Spin-Axis Orientation on Thermal Control of Large Probe	5-33
5-20	Orbiter Antenna Tradeoffs	5-36
5-21	Comparison Between Power Outputs of Conical and Cylindrical Arrays	5-37

..CEDING PAGE BLANK NOT FILMED

ILLUSTRATIONS (CONTINUED)

		Page
5-22	Recommended Thor/Delta Probe Bus	5-41
5-23	Thor/Delta Probe Mission Science and Equipment Layout	5-42
5-24	Atlas/Centaur Probe Mission Science and Equipment Layout With Other Candidate Instruments	5-43
5-25	Preferred Atlas/Centaur Probe Mission Spacecraft	5-45
5-26	Existing DSCS-II is Costly to Adapt even for Simultaneous Release of Small Probes and would be even more Costly to Adapt for Sequential Release	5-46
5-27	Thor/Delta Orbiter Despun Reflector and Fanscan	5-49
5-28	Thor/Delta Orbiter 12-Watt Fanbeam/Fanscan	5-50
5-29	Thor/Delta Orbiter 36-Watt Fanbeam/Fanscan	5-52
5-30	Thor/Delta Orbiter Mission Spacecraft Normal-to-Venus Orbital Plane Electronic Despun Antenna	5-53
5-31	Thor/Delta Earth Pointer	5-54
5-32	Preferred Atlas/Centaur Orbiter Mission Earth-Pointing Spacecraft, Version IV Science Payload	5-56
5-33	Atlas/Centaur Orbiter Option 2 Despun Antenna	5-47
5-34	Atlas/Centaur Orbiter Option 2, 3-1Watt Fanbeam/Fanscan	5-58
5-35	Atlas/Centaur Orbiter Option 3, 12-Watt Fanbeam/Fanscan	5-59
5-36	Preferred Atlas/Centaur Orbiter Configuration	5-61
5-37	Thor/Delta Mission Spacecraft ESRO Propulsion and Antenna	5-62
5-38	Thor/Delta Probe Mission Spacecraft ESRO Propulsion	5-63
5-39	Thor/Delta Launch Configuration	5-65
5-40	Atlas/Centaur Launch Configuration	5-64
5-41	Typical Response Spectra Envelopes for Spacecraft Pyrotechnic Release Shock (5 Percent Damping)	5-68
6-1	Atlas/Centaur Descent Capsule, Large and Small Probes	6-2
6-2	Large Probe Entry Trajectory Parameters	6-3
6-3	Large Probe Performance Requirements	6-5
6-4	Small Probe Performance/Requirements	6-6
6-5	Large Probe/Probe Bus Mechanical Interface	6-8
6-6	Small Probe/Probe Bus Mechanical Interface	6-9
6-7	Small Probe/Probe Bus Release Mechanism	6-9

ILLUSTRATIONS (CONTINUED)

		Page
6-8	Atlas/Centaur Large Probe Reference Coordinates	6-13
6-9	Atlas/Centaur Small Probe Reference Coordinates	6-14
6-10	Large Probe Functional Block Diagram, Atlas/Centaur	6-17
6-11	Small Probe Functional Block Diagram, Atlas/Centaur	6-18
6-12	Probe Mission Profile	6-22
6-13	Thor/Delta Large Probe After Aeroshell Release	6-26
6-14	Thor/Delta Small Probe	6-26
6-15	Thor/Delta Small Probe Integrated Electronics	6-27
6-16	Thor/Delta Large Probe Reference Coordinates	6-31
6-17	Thor/Delta Small Probe Reference Coordinates	6-31
6-18	Large Probe Functional Diagram, Thor/Delta	6-33
6-19	Small Probe Functional Diagram, Thor Delta	6-34
6-20	Thor/Delta Power Transfer Schematic	6-36
6-21	Communications Subsystem, Thor/Delta Large Probe	6-38
6-22	Communications Subsystem, Thor/Delta Small Probe	6-38
6-23	Weight Evaluation Milestone	6-41
6-24	Weight Margin Depletion Schedule	6-42
6-25	Exploded View of Orbiter and Probe Bus	6-46
6-26	Simplicity and Commonality of Structural Design	6-47
6-27	Attitude and Velocity Control Thruster Arrangement	6-49
6-28	Preferred Earth-Pointing Atlas/Centaur Spacecraft Mass Properties Summary, Version IV Science Payload and 1978 Mission Launches	6-53
6-29	Preferred Fanbeam, Fanscan Atlas/Centaur Spacecraft Mass Properties Summary for Version III Science Payload and 1977/1978 Mission Launches	6-57
6-30	Probe Bus Functional Block Diagram, Preferred Atlas/Centaur Configuration	6-61
6-31	Orbiter Functional Block Diagram, Preferred Atlas/Centaur Configuration	6-62
6-32	Conical Solar Array Characteristics	6-63
6-33	Probe Bus Antenna Pattern Coverage	6-64
6-34	Orbiter Antenna Pattern Coverage	6-66
6-35	Heliocentric Plan View of Pioneer Venus Orbiter Trajectory Showing Telemetry Bit Rate Capability	6-67

ILLUSTRATIONS (CONTINUED)

		Page
6-36	Doppler Modulation/Shift Technique for Attitude Determination	6-68
6-37	Conscan Method for Attitude Determination	6-69
6-38	Atlas/Centaur Probe Bus Electrical Distribution Diagram	6-73
6-39	Atlas/Centaur Orbiter Electrical Distribution Diagram	6-74
6-40	Exploded View of Thor/Delta Probe Bus Spacecraft	6-77
6-41	Simplicity and Commonality of Structural Design	6-78
6-42	Example of Angle of Attack Trace in Inertial Space (Thor/Delta Probe, First Midcourse Maneuver)	6-79
6-43	Examples of Amplitudes of Angle of Attack, Nutation Angle, and Momentum Vector Attitude Versus Time (Thor/Delta Probe, First Midcourse Maneuver)	6-79
6-44	Preferred Fanbeam, Fanscan Thor/Delta Spacecraft Mass Properties Summary for Version III Science Payload and 1977/1978 Mission Launches	6-83
6-45	Probe Bus Functional Block Diagram, Preferred Thor/Delta Configuration	6-87
6-46	Orbiter Functional Block Diagram, Preferred Thor/Delta Configuration	6-88
6-47	Thor/Delta Power Subsystem	6-89
6-48	Functional Arrangement of Antennas, Preferred Thor/Delta Configuration	6-89
6-49	Data Storage Concept, Preferred Thor/Delta Configuration	6-90
6-50	Atlas/Centaur Probe Mission Reliability Diagram (Probe Bus and Large Probe Mission Critical)	6-94
6-51	Atlas/Centaur Probe Mission Reliability Diagram (Probe Bus Only Mission Critical)	6-94
6-52	Thor/Delta Reliability Studies	6-95
6-53	Atlas/Centaur Reliability Studies	6-96
6-54	Mission Objective Data Contribution Large Probe/ Small Probe	6-100
6-55	Reliability/Weight Tradeoff Curve for Thor/Delta Probe Bus	6-103
6-56	Thor/Delta Probe Bus Reliabilty Block Diagram	6-105
6-57	Key Reliability Features	6-106
6-58	Atlas/Centaur Probe Bus Reliability Block Diagram	6-107
6-59	Thor/Delta Orbiter Reliability Block Diagram	6-109
6-60	Atlas/Centaur Orbiter Reliability Block Diagram	6-111

ACRONYMS AND ABBREVIATIONS

A	ampere analog
abA	abampere
AC	alternating current
A/C	Atlas/Centaur
ADA	avalanche diode amplifier
ADCS	attitude determination and control subsystem
ADPE	automatic data processing equipment
AEHS	advanced entry heating simulator
AEO	aureole/extinction detector
AEDC	Arnold Engineering Development Corporation
AF	audio frequency
AGC	automatic gain control
AgCd	silver-cadmium
AgO	silver oxide
AgZn	silver zinc
ALU	authorized limited usage
AM	amplitude modulation
a.m.	ante meridian
AMP	amplifier
APM	assistant project manager
ARC	Ames Research Center
ARO	after receipt of order
ASK	amplitude shift key
at. wt	atomic weight
ATM	atmosphere
ATRS	attenuated total refractance spectrometer
AU	astronomical unit
AWG	American wire gauge
AWGN	additive white gaussian noise
B	bilevel
B	bus (probe bus)
BED	bus entry degradation

BER	bit error rate
BLIMP	boundary layer integral matrix procedure
BPIS	bus-probe interface simulator
BPL	bandpass limiter
BPN	boron potassium nitrate
bps	bits per second
BTU	British thermal unit
C	Canberra tracking station – NASA DSN
CADM	configuration administration and data management
C&CO	calibration and checkout
CCU	central control unit
CDU	command distribution unit
CEA	control electronics assembly
CFA	crossed field amplifier
cg	centigram
c.g.	center of gravity
CIA	counting/integration assembly
CKAFS	Cape Kennedy Air Force Station
cm	centimeter
c.m.	center of mass
C/M	current monitor
CMD	command
CMO	configuration management office
C-MOS	complementary metal oxide silicon
CMS	configuration management system
const	constant construction
COSMOS	complementary metal oxide silicon
c.p.	center of pressure
CPSA	cloud particle size analyzer
CPSS	cloud particle size spectrometer

ACRONYMS AND ABBREVIATIONS (CONTINUED)

CPU	central processing unit
CRT	cathode ray tube
CSU	Colorado State University
CTRF	central transformer rectifier filter
D	digital
DACS	data and command subsystem
DCE	despin control electronics
DDA	despin drive assembly
DDE	despin drive electronics
DDU	digital decoder unit
DDULBI	doubly differenced very long baseline interferometry
DEA	despin electronics assembly
DEHP	di-2-ethylhexyl phthalate
DFG	data format generator
DGB	disk gap band
DHC	data handling and command
DIO	direct input/output
DIOC	direct input/output channel
DIP	dual in-line package
DISS REG	dissipative regulator
DLA	declination of the launch azimuth
DLBI	doubly differenced very long baseline interferometry
DMA	despin mechanical assembly
DOF	degree of freedom
DR	design review
DSCS II	Defense System Communications Satellite II
DSIF	Deep Space Instrumentation Facility
DSL	duration and steering logic
DSN	NASA Deep Space Network
DSP	Defense Support Program
DSU	digital storage unit
DTC	design to cost
DTM	decelerator test model

DTP	descent timer/programmer
DTU	digital telemetry unit
DVU	design verification unit
E	encounter entry
EDA	electronically despun antenna
EGSE	electrical ground support equipment
EIRP	effective isotropic radiated power
EMC	electromagnetic compatibility
EMI	electromagnetic interference
EO	engineering order
EOF	end of frame
EOM	end of mission
EP	earth pointer
ESA	elastomeric silicone ablator
ESLE	equivalent station error level
ESRO	European Space Research Organization
ETM	electrical test model
ETR	Eastern Test Range
EXP	experiment
FFT	fast Fourier transform
FIPP	fabrication/inspection process procedure
FMEA	failure mode and effects analysis
FOV	field of view
FP	fixed price frame pulse (telemetry)
FS	federal stock
FSK	frequency shift keying
FTA	fixed time of arrival

G	Goldstone Tracking Station - NASA DSN gravitational acceleration
g	gravity
G&A	general and administrative
GCC	ground control console
GFE	government furnished equipment
GHE	ground handling equipment
GMT	Greenwich mean time
GSE	ground support equipment
GSFC	Goddard Space Flight Center
H	Haystack Tracking Station - NASA DSN
HFFB	Ames Hypersonic Free Flight Ballistic Range
HPBW	half-power beamwidth
htr	heater
HTT	heat transfer tunnel
I	current
IA	inverter assembly
IC	integrated circuit
ICD	interface control document
IEEE	Institute of Electrical and Electronics Engineering
IFC	interface control document
IFJ	in-flight jumper
IMP	interplanetary monitoring platform
I/O	input/output
IOP	input/output processor
IR	infrared
IRAD	independent research and development
IRIS	infrared interferometer spectrometer
IST	integrated system test
I&T	integration and test
I-V	current-voltage

JPL	Jet Propulsion Laboratory
KSC	Kennedy Space Center
L	launch
LD/AD	launch date/arrival date
LP	large probe
LPM	lines per minute
LPTTL MSI	low power transistor-transistor logic medium scale integration
LRC	Langley Research Center
M	Madrid tracking station - NASA DSN
MAG	magnetometer
max	maximum
MEOP	maximum expected operating pressure
MFSK	M'ary frequency shift keying
MGSE	mechanical ground support equipment
MH	mechanical handling
MIC	microwave integrated circuit
min	minimum
MJS	Mariner Jupiter-Saturn
MMBPS	multimission bipropellant propulsion subsystem
MMC	Martin Marietta Corporation
MN	Mach number
mod	modulation
MOI	moment of inertia
MOS LSI	metal over silicone large scale integration
MP	maximum power
MSFC	Marshall Space Flight Center
MPSK	M'ary phase shift keying
MSI	medium scale integration
MUX	multiplexer
MVM	Mariner Venus-Mars

NAD	Naval Ammunition Depot, Crane, Indiana
N/A	not available
NiCd	nickel cadmium
NM/IM	neutral mass spectrometer and ion mass spectrometer
NRZ	non-return to zero
NVOP	normal to Venus orbital plane
OEM	other equipment manufacturers
OGO	Orbiting Geophysical Observatory
OIM	orbit insertion motor
P	power
PAM	pulse amplitude modulation
PC	printed circuit
PCM	pulse code modulation
PCM-PSK-PM	pulse code modulation-phase shift keying-phase modulation
PCU	power control unit
PDA	platform drive assembly
PDM	pulse duration modulation
PI	principal investigator proposed instrument
PJU	Pioneer Jupiter-Uranus
PLL	phase-locked loop
PM	phase modulation
p.m.	post meridian
P-MOS	positive channel metal oxide silicon
PMP	parts, materials, processes
PMS	probe mission spacecraft
PMT	photomultiplier tube
PPM	parts per million pulse position modulation
PR	process requirements
PROM	programmable read-only memory
PSE	program storage and execution assembly

ACRONYMS AND ABBREVIATIONS (CONTINUED)

PSIA	pounds per square inch absolute
PSK	phase shift key
PSU	Pioneer Saturn-Uranus
PTE	probe test equipment
QOI	quality operation instructions
QTM	qualification test model
RCS	reaction control subsystem
REF	reference
RF	radio frequency
RHCP	right hand circularly polarized
RHS	reflecting heat shield
RMP-B	Reentry Measurements Program, Phase B
RMS	root mean square
RMU	remote multiplexer unit
ROM	read only memory rough order of magnitude
RSS	root sum square
RT	retargeting
RTU	remote terminal unit
S	separation
SBASI	single bridgewire Apollo standard initiator
SCP	stored command programmer
SCR	silicon controlled rectifier
SCT	spin control thrusters
SEA	shunt electronics assembly
SFOF	Space Flight Operations Facility
SGLS	space ground link subsystem
SHIV	shock induced vorticity
SLR	shock layer radiometer
SLRC	shock layer radiometer calibration

SMAA	semimajor axis
SMIA	semiminor axis
SNR	signal to noise ratio
SP	small probe
SPC	sensor and power control
SPSG	spin sector generator
SR	shunt radiator
SRM	solid rocket motor
SSG	Science Steering Group
SSI	small scale integration
STM	structural test model
STM/TTM	structural test model/thermal test model
STS	system test set
sync	synchronous
TBD	to be determined
TCC	test conductor's console
T/D	Thor/Delta
TDC	telemetry data console
TEMP	temperature
TS	test set
TTL MSI	transistor-transistor logic medium scale integration
TLM	telemetry
TOF	time of flight
TRF	tuned radio frequency
TTM	thermal test model
T/V	thermo vacuum
TWT	travelling wave tube
TWTA	travelling wave tube amplifier
UHF	ultrahigh frequency
UV	ultraviolet

ACRONYMS AND ABBREVIATIONS (CONTINUED)

VAC	volts alternating current
VCM	vacuum condensable matter
VCO	voltage controlled oscillator
VDC	volts direct current
VLBI	very long baseline interferometry
VOI	Venus orbit insertion
VOP	Venus orbital plane
VSI	Viking standard initiator
VTA	variable time of arrival
XDS	Xerox Data Systems

5. System Configuration Concepts and Tradeoffs

5.1 Probe Configurations and Tradeoffs

5. SYSTEM CONFIGURATION CONCEPTS AND TRADEOFFS

It was recognized early in the study that the weight and volume restraints imposed by the Thor/Delta launch vehicle would require costly probe development and test effort. The probe studies have shown, however, that viable configurations can be developed for the Thor/Delta. The Version III science payload (nominal and additional instruments except for the small probe magnetometers) could be accommodated, but the margins were tight, and it was shown in the midterm presentation that it would be very expensive to allocate a few more kilograms for the probes.

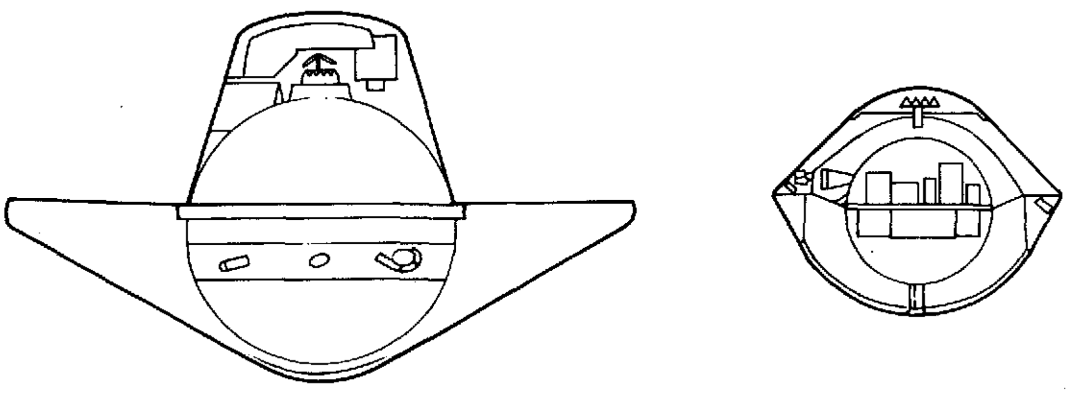
Significant cost savings can be realized by designing the probes for the Atlas/Centaur. Increased probe weight allocation permits increased design margins, reduced risk, and greater use of flight-proven hardware.

Figure 5-1 shows the preferred Atlas/Centaur probe designs, summarizes the comparisons between the Thor/Delta and Atlas/Centaur versions, and indicates results of major tradeoff studies. Even before the selection of the Atlas/Centaur launch vehicles, it was shown that the use of Atlas/Centaur for the probe mission would be by far the lowest-cost, lowest-risk selection. The Version IV science payload only reinforces that choice. Section 5.1 shows, in some detail, how the probe configurations evolved during the study as a result of the tradeoffs that were performed.

The probe configurations have a major impact on the design of the probe bus, which must be designed primarily to protect the probes in transit to Venus and launch them accurately. Probe design also indirectly impacts the orbiter design because of the need for commonality between the two mission spacecraft as a means of minimizing total program cost.

The most significant cost/performance tradeoffs for the bus and orbiter stemmed from the selection of a spin axis orientation for: 1) the bus, to provide large-probe thermal protection without the need for jettisonable covers or heaters, and 2) the orbiter, to permit use of a fixed, high-gain antenna to accommodate the specified science data rate. Before the upward revision of data requirements for the Version IV

PREFERRED ATLAS/CENTAUR PROBE CONFIGURATIONS



LARGE PROBE CHARACTERISTICS

- 2.09- RAD (120-DEG) BLUNT CONE FOREBODY, MINIMUM VOLUME AFTERBODY
- SINGLE STAGING WITH PILOT-DEPLOYED MAIN PARACHUTE
- DESCENT CAPSULE STABILIZED BY PERFORATED RING
- PRESSURE-PROOF, EXTERNALLY INSULATED, ALUMINUM DESCENT CAPSULE
- COMMONALITY OF SUBSYSTEM HARDWARE WITH SMALL PROBES
- AEROSHELL DIAMETER 28 PERCENT GREATER THAN THOR/DELTA
- INCREASED STRUCTURAL, THERMAL, HEAT SHIELD MARGINS
- PRESSURE VESSEL VOLUME INCREASED 30 PERCENT OVER THOR/DELTA

SMALL PROBE CHARACTERISTICS

- 1.57-RAD (90-DEG) BLUNT CONE FOREBODY, CONE/SPHERICAL CAP AFTERBODY
- UNSTAGED
- AERODYNAMICALLY STABLE, HYPERSONIC TO SUBSONIC
- SENSORS EXPOSED AFTER ENTRY HEATING AND LOADING
- EXTERNALLY INSULATED ALUMINUM PRESSURE VESSEL
- AEROSHELL DIAMETER 73 PERCENT GREATER THAN THOR/DELTA
- INCREASED SAFETY FACTORS
- PRESSURE VESSEL VOLUME INCREASED 300 PERCENT
- HIGHER TRANSMITTER POWER THAN THOR/DELTA VERSION

MAJOR PROBE TRADEOFFS

1. AEROSHELL SHAPE

BLUNTED CONE PROVIDES LARGE DRAG FOR HIGH-ALTITUDE DECELERATIONS, PERMITTING LARGE PROBE TO REACH SUBSONIC VELOCITIES ABOVE MAIN CLOUD LAYERS (70 KM). AERODYNAMICALLY STABLE HYPERSONICALLY AND SUBSONICIALLY, WITH LOW ANGLE OF ATTACK AND QUICKLY TRAVERSED TRANSONIC INSTABILITY.

LESS BLUNT CONFIGURATION THAN LARGE PROBE PERMITS PACKAGING INSTRUMENT CAPSULE FURTHER FORWARD, THUS IMPROVING AERODYNAMIC STABILITY IN ALL SPEED RANGES, AND PROVIDES LOWER SHOCK LAYER RADIATIVE HEATING. THIS CONFIGURATION PROVIDES SUFFICIENT DRAG FOR DECELERATION TO LOW SUPERSONIC SPEEDS ABOVE MAIN CLOUDS.

2. STAGED VERSUS UNSTAGED, STAGING TECHNIQUE

STAGING PERMITS RELIABLE, UNCONTAMINATED VIEWING AND SAMPLING BY SCIENCE INSTRUMENTS; SHORTENS DESCENT TIME IN COMPARISON WITH UNSTAGED VERSION, REDUCING BATTERY AND THERMAL CONTROL WEIGHT. USE OF PARACHUTE TO STAGE DESCENT CAPSULE OUT OF AEROSHELL IS LIGHTER, LESS EXPENSIVE, MORE RELIABLE THAN MECHANICAL METHODS. USE OF PILOT CHUTE PERMITS SIMPLER INTEGRATION OF MAIN CHUTE IN PROBE.

SIZE AND WEIGHT LIMITATIONS RULE OUT SMALL PROBE STAGING. SCIENCE INSTRUMENTS ARE DEPLOYED DIRECTLY THROUGH AEROSHELL BY RELIABLE NONPYROTECHNIC DEVICES AFTER PROBES DECELERATE TO APPROXIMATELY MACH 1.5 AT 70 KM.

3. DESCENT PROFILE

THE SELECTED BALLISTIC COEFFICIENTS OF DESCENT CAPSULE-WITH-CHUTE FOLLOWED BY DESCENT CAPSULE ALONE (AFTER CHUTE JETTISON AT 43 KM) PROVIDE A DESCENT PROFILE FULLY COMPATIBLE WITH SCIENCE SAMPLING REQUIREMENTS. BATTERY AND THERMAL CONTROL WEIGHT CAN BE SAVED BY USING HIGHER COEFFICIENTS AND BY STAGING AT HIGHER ALTITUDE BUT THERE IS SOME SACRIFICE IN SCIENCE DATA. DATA ACQUISITION RATE AND MEMORY SIZE ARE EASILY IMPLEMENTED WITH MODIFIED PIONEERS 10 AND 11 HARDWARE.

AEROSHELL GEOMETRY YIELDS LOW TRANSONIC AND SUBSONIC BALLISTIC COEFFICIENTS. DESCENT VELOCITY THRU IMPORTANT CLOUD LAYERS IS SUFFICIENTLY LOW TO SATISFY SCIENCE SAMPLING REQUIREMENTS. TOTAL DESCENT TIME OF 65 MINUTES IS NEARLY OPTIMUM COMPROMISE BETWEEN SCIENCE DATA RETURN AND BATTERY/THERMAL CONTROL WEIGHT REQUIREMENTS.

4. TERMINAL DESCENT STABILITY

EXTENSIVE SPIN TUNNEL TESTING OF FINS, FLARES, AND DRAG RINGS ON SPHERICAL PRESSURE VESSEL LED TO SELECTION OF AN EQUATORIAL RING AS THE PREFERRED STABILIZING DEVICE. CANTED HOLES IN RING INDUCE APPROPRIATE SPIN RATE, ELIMINATING NEED FOR FINS. DESCENT CAPSULE/RING GEOMETRY PERMITS ATTACHMENT NEAR PROBE CENTER OF GRAVITY AND PERMITS AEROSHELL AFTERBODY OF MINIMUM VOLUME.

SELECTED AEROSHELL AFTERBODY IS BASED ON TEST OF NUMEROUS SHAPES. IT PROVIDES ADEQUATE SUBSONIC STABILITY WITH ACHIEVABLE CENTER OF GRAVITY LOCATION, MINIMIZES LIMIT CYCLE BEHAVIOR, AND HAS CAPABILITY TO RECOVER RAPIDLY FROM MAXIMUM ANGLE OF ATTACK PERTURBATIONS RESULTING FROM SPECIFIED WIND PROFILES.

Figure 5-1. Probe Configuration and Tradeoff Conclusions

science payload, another less expensive orbiter configuration was preferred but the new higher data rate made it nonviable.

Other orbiter configuration options were also studied under the original guidelines. In particular, a set of alternative configurations derived from the existing DSCS-II spacecraft (which has a despin antenna) were examined, but the number of modifications required to adapt the design to the Venus missions and the requirement (for these configurations) to launch the small probes simultaneously (as opposed to the recommended sequential release) make these alternative configurations noncompetitive.

Figure 5-2 shows the major configurations studied and the results of the major configuration tradeoffs. Section 5.2 describes the probe bus and orbiter configuration tradeoffs and the designs derived for both launch vehicle candidates.

5.1 PROBE CONFIGURATIONS AND TRADEOFFS

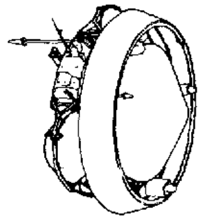
The preferred probe configurations finalized at the conclusion of this study are summarized in Section 5.1.1 and are further defined in Sections 6 and 7. These configurations use the Atlas/Centaur as the specified launch vehicle for a 1978 launch, and carry the Version IV science payload. The major configuration options used to derive the preferred probe concepts are developed in Section 5.1.2. Many of these options were also the key elements varied during the pre-midterm studies.

Tables 5-1A and 5-1B summarize the system and science instrument requirements of the Phase B study defined by the NASA/ARC documents referenced in these tables. From the system design study requirements and the science instrument definitions provided, the probe mission and system design requirements (summarized in Tables 5-1C, 5-1D, and 5-2) were derived to satisfy the Pioneer Venus project objectives. The preferred configurations for the large and small probes meet these objectives and are the result of an intense and comprehensive effort to provide a low-cost, low-risk program.

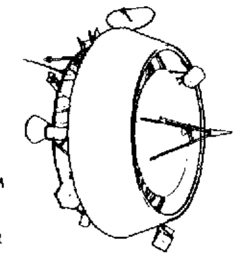
In arriving at the preferred probe designs, a number of configurations were evaluated against the following criteria: cost, performance, weight,

A PREFERRED AND ALTERNATIVE PROBE BUS AND ORBITER CONFIGURATION TRADEOFFS

PREFERRED CONFIGURATIONS



PREFERRED PROBE BUS	PREFERRED ORBITER
EARTH-POINTING CONICAL SOLAR ARRAY SEQUENTIAL RELEASE 6.23 RAD (360-DEG) OMNI'S PLUS MEDIUM GAIN HORN	EARTH-POINTING CONICAL SOLAR ARRAY FIXED HIGH-GAIN DISH W/CONSCAN 6-WATT TRANSMITTER GIMBALED RAM PLATFORM

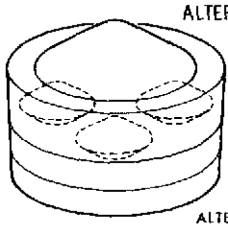


BOTH BUS AND ORBITER DESIGNS LAUNCHED BY ATLAS/CENTAUR

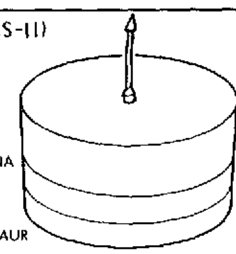
DESIGN DRIVERS

PROBE BUS	ORBITER
1. SPIN AXIS ORIENTATION: EARTH-POINTING OR NORMAL TO VENUS ORBIT PLANE	
WITH EARTH-POINTING SPIN AXIS, FAVORABLE SUN ANGLE PERMITS USE OF SOLAR RADIATION FOR LARGE PROBE THERMAL CONTROL INSTEAD OF HEATERS (AS LONG AS SOLAR ARRAY IS CONICAL) SO THE BATTERY AND ARRAY CAN BE SMALLER THAN IT WOULD BE WITH SPIN AXIS PERPENDICULAR AND CYLINDRICAL ARRAY.	WITH ORBITER SPIN AXIS NORMAL TO VENUS ORBIT PLANE, RAM INSTRUMENTS NEED NO POINTABLE GIMBALLED PLATFORM
2. ANTENNA CONFIGURATION	
PROBE BUS HAS 6.23 RAD (360-DEG) OMNI COVERAGE PLUS PIONEERS 10 AND 11 HORN FOR HIGH BIT RATE DATA TRANSMISSION DURING ENTRY (1024 BITS/S).	PREFERRED ORBITER HAS FIXED, HIGH-GAIN ANTENNA AND ADEQUATE BIT RATE WITH EXTREMELY HIGH RELIABILITY I.E., NO SLIP RINGS, ROTARY COUPLERS, OR OTHER COMPLEXITIES).
3. SOLAR ARRAY: CONICAL OR CYLINDRICAL	
CONICAL ARRAY ON BUS PERMITS MANEUVERING WITHOUT HASTE AND PERMITS USE OF SOLAR RADIATION TO KEEP LARGE PROBE WARM WITHOUT HEATERS. LOWER PEAK POWER THAN AN EQUIVALENT CYLINDRICAL ARRAY, BUT HIGHER AVERAGE POWER THROUGH WIDE RANGE OF MANEUVERS, SO BACKUP BATTERY CAN BE SMALLER THAN WITH CYLINDRICAL ARRAY. ELIMINATION OF NEED FOR HEATERS ALSO ALLOWS SMALLER ARRAY.	CONICAL ARRAY ON ORBITER IS NOT AS ADVANTAGEOUS AS FOR BUS DURING TRANSIT, BUT OFFERS GREAT ADVANTAGES IN ORBIT BECAUSE IT PERMITS WIDE RANGE OF MANEUVERS WITH SMALLER BACKUP BATTERY REQUIREMENTS.
4. SEQUENTIAL VERSUS SIMULTANEOUS PROBE RELEASE	
SEQUENTIAL RELEASE PERMITS CONTROLLED SEPARATION OF RELEASE TIMES SO THAT PROBES CAN BE CAREFULLY TARGETED FOR ENTRY INTO VENUS ATMOSPHERE AT ZERO ANGLE OF ATTACK (TO ENHANCE UPPER ATMOSPHERE DENSITY MEASUREMENTS BASED ON ACCELEROMETER DATA) AND FOR GOOD GEOGRAPHICAL DISPERSION OF PROBE ENTRY POINTS. THIS ALSO FACILTATES REDUNDANT COMMUNICATIONS COVERAGE BY THE DSN.	

ALTERNATIVE CONFIGURATIONS (Derived from existing DSCS-II)

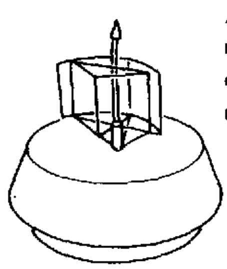


ALTERNATIVE PROBE BUS	ALTERNATIVE ORBITER
NORMAL TO EARTH LINE CYLINDRICAL ARRAY SIMULTANEOUS RELEASE MODIFIED DSCS-II	NORMAL TO EARTH LINE CYLINDRICAL ARRAY FIXED FANSCAN ANTENNA MODIFIED DSCS-II



ALTERNATIVE DESIGNS CAN ONLY BE LAUNCHED BY ATLAS/CENTAUR

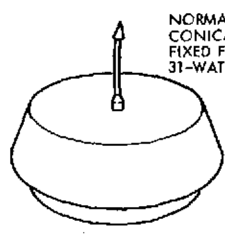
B ORBITER OPTIONS - Any of these configurations can be launched by either Thor/Delta or Atlas/Centaur



ORBITER OPTION 1

NORMAL TO EARTH LINE
CONICAL SOLAR ARRAY
DESPUN HIGH-GAIN REFLECTOR

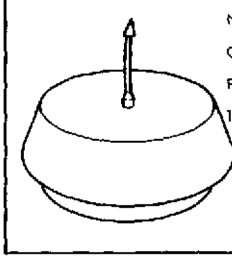
OPTION 1 IS COMPATIBLE WITH ESRO PARTICIPATION BUT INCREASES COST CONSIDERABLY AND REDUCES RELIABILITY.



ORBITER OPTION 2

NORMAL TO EARTH LINE
CONICAL SOLAR ARRAY
FIXED FANSCAN ANTENNA
31-WATT TRANSMITTER

OPTION 2 IS ONLY VIABLE FOR THE INITIAL STUDY GUIDELINES. ECONOMICAL, MEDIUM-GAIN ANTENNA AND HIGH-POWER TRANSMITTER MEET DATA RATE REQUIREMENTS WITH 64 AND 26 METER DSN SUBNETS



ORBITER OPTION 3

NORMAL TO EARTH LINE
CONICAL SOLAR ARRAY
FIXED FANSCAN ANTENNA
12-WATT TRANSMITTER

OPTION 3 SAVES MONEY WITH LOWER POWER TRANSMITTER BUT CANNOT USE 26 METER DSN SUBNET FOR BACKUP. VIABLE ONLY FOR INITIAL STUDY GUIDELINES

Figure 5-2. Probe Bus and Orbiter Configurations and Tradeoffs Conclusions

Table 5-1A. System Design Study Requirements Summary

SCIENCE MISSION OBJECTIVES	NATURE/COMPOSITION OF CLOUDS COMPOSITION/STRUCTURE OF ATMOSPHERE TO SURFACE GENERAL CIRCULATION PATTERN OF LOWER ATMOSPHERE CURSORY INVESTIGATION OF COMPOSITION/STRUCTURE OF IONOSPHERE
LAUNCH VEHICLE	ATLAS/CENTAUR (SELECTED) THOR/DELTA (COMPARISON)
ENVIRONMENT	NASA SP-8011 MODELS OF VENUS ATMOSPHERE
MULTIPLE-PROBE BASELINE CONFIGURATION	1 - BUS 1 - LARGE PROBE 3 - IDENTICAL SMALL PROBES
COMMONALITY	MAXIMUM SYSTEM/SUBSYSTEM BETWEEN BUS AND ORBITER
DEEP SPACE NETWORK	TELEMETRY, COMMAND, TRACKING COMPATIBLE WITH 64 M/26 M DSIF NETWORKS, SFOF, AND GCF
SPACECRAFT STABILIZATION	SPIN STABILIZED
COMMUNICATION PERFORMANCE	COMMAND LINK BIT ERROR PROBABILITY NO GREATER THAN 10^{-5} TELEMETRY LINK UNDETECTED BIT ERROR PROBABILITY NO GREATER THAN 10^{-3} AND DATA WORD DELETION NO GREATER THAN 10^{-2}
LAUNCH SITE	CKAFS
BASELINE INSTRUMENT PAYLOAD	NASA-ARC LETTER ASD 244-9/32-042, 13 APRIL 1973 (VERSION IV)
MAGNETIC PROPERTIES	CONSISTENT WITH SCIENTIFIC OBJECTIVES
HEAT SHIELD	DEMONSTRATED RELIABILITY ON-SITE REPAIRABILITY - RELIABLE SURVIVAL THROUGH ALL PHASES, NON-INTERFERRING WITH SCIENCE, RF TRANSPARENT KNOWN FABRICATION/BONDING PROPERTIES, ADEQUATE SHELF LIFE EXPERIENCE, NONDESTRUCTIVE TESTING
PROBE AERODYNAMICS	EXTERNAL FORM/INTERNAL MASS DISTRIBUTION FOR PROPER FLIGHT CHARACTERISTICS THROUGH ENTRY, PARACHUTE DESCENT, AND FREE DESCENT

Table 5-1B. Large and Small Probe Nominal Science Instrument Requirements Summary

LARGE PROBE		
WEIGHT	NOMINAL: 27.17 KG (60.45 LB)	+ 15% = 31.25 KG (69.52 LB) - 10% = 24.45 KG (54.41 LB)
POWER	NOMINAL: 88.6 WATTS AVERAGE	+ 20% = 106.3 WATTS - 10% = 79.7 WATTS
VOLUME	29 810 CC (1823 $IN.^3$)	+ 15% = 34 281 CC (2096 $IN.^3$) - 15% = 25 338 CC (1550 $IN.^3$)
DATA	SATISFY SPECIFIED MINIMUM SAMPLING INTERVALS MASS SPECTROMETER 80 000 BITS/66 TO 44 KM AND 88 000 BITS/44 KM TO SURFACE GAS CHROMATOGRAPH 13 200 BITS BUFFER MEMORY, 10 MINUTE STORE/READOUT TIME, THREE 20 MINUTE MEASUREMENT CYCLES ACCELEROMETER: 1000 BITS STORED DURING ENTRY	
PENETRATIONS	WINDOWS: 3 (SOLAR RADIOMETER - CPSA - IR FLUX RADIOMETER) INLETS: 3 (PRESSURE GAUGE - NEUTRAL MASS SPECTROMETER - GAS CHROMATOGRAPH) OTHER: 3 (TEMPERATURE GAUGE - HYGROMETER - WIND ALTIMETER RADAR)	
ALTITUDE	ACCELEROMETER 140 KM (4 X 10^{-4} G) TO SURFACE HYGROMETER 70 KM TO 40 KM WIND ALTITUDE RADAR 40 KM TO SURFACE ALL OTHERS 70 KM TO SURFACE	
SPIN/STABILITY	SOLAR AND IR FLUX RADIOMETERS: 1 TO 5 RPM SOLAR RADIOMETER: ANGLE WITH RESPECT TO LOCAL VERTICAL KNOWN TO WITHIN $\pm 2^\circ$	
SMALL PROBE		
WEIGHT	NOMINAL: 2.17 KG (4.9 LB)	+ 15% = 2.5 KG (5.64 LB) - 5% = 2.06 KG (4.66 LB)
POWER	NOMINAL: 4.25 WATTS AVERAGE	+ 20% = 5.1 WATTS - 10% = 3.83 WATTS
VOLUME	1231 CC (75 $IN.^3$)	+ 15% = 1416 CC (86 $IN.^3$) - 15% = 1046 CC (64 $IN.^3$)
DATA	SATISFY SPECIFIED MINIMUM SAMPLING INTERVALS FROM 66 KM TO SURFACE ACCELEROMETER: 250 BITS STORED DURING ENTRY	
PENETRATIONS	WINDOWS: 3 (NEPHELOMETER (2) - IR FLUX DETECTOR) INLET: 1 (PRESSURE GAUGE) OTHER: 1 (TEMPERATURE GAUGE)	
ALTITUDE	ACCELEROMETER 140 KM (4 X 10^{-4} G) TO SURFACE	
SPIN/STABILITY	NOT DEFINED	

Table 5-1C. Probe Interface Design Requirement Summary

SCIENCE INTERFACE		
LARGE PROBE	10 SCIENTIFIC INSTRUMENTS 31.25 KG TOTAL WEIGHT (MAX) MOUNTING - FLAT WITHIN 0.005 IN. ALIGNMENT - $\pm 0.5^\circ$ ELECTRICAL POWER - 106.3 WATTS MAX @ 28 VDC ± 10% TIMING - SEQUENCING SIGNALS TELEMETRY SIGNALS - ANALOG/DIGITAL/BILEVEL THERMAL - 0 TO 150°F	
SMALL PROBE	6 SCIENTIFIC INSTRUMENTS 2.5 KG TOTAL WEIGHT (MAX) MOUNTING - FLAT WITHIN 0.005 IN. ALIGNMENT - $\pm 0.5^\circ$ ELECTRICAL POWER - 5.1 WATTS @ +28 VDC ± 10% TIMING - SEQUENCING SIGNALS TELEMETRY SIGNALS - ANALOG/DIGITAL/BILEVEL THERMAL - 0 TO 150°F	
PROBE BUS INTERFACE		
MASS		
MASS PROPERTIES		
LARGE PROBE -	263.6 KG (581.2 LB) + 15%	
CG TOLERANCE -	RADIAL C.G. OFFSET FROM PROBE CENTERLINE $\le$ 0.050 IN. AND LONGITUDINAL UNCERTAINTY FROM NOMINAL $\le$ 0.030 IN.	
PRINCIPAL AXIS TOLERANCE =	$\pm$ 100 LBM-$IN.^2$ UNCERTAINTY IN BALANCE OF PROBE	
SMALL PROBE -	70.0 KG (154.4 LB) + 15%	
CG TOLERANCE =	RADIAL C.G. OFFSET FROM PROBE CENTERLINE $\le$ 0.050 IN. AND LONGITUDINAL UNCERTAINTY FROM NOMINAL $\le$ 0.050 IN.	
PRINCIPAL AXIS TOLERANCE =	$\pm$ 35 LBM-$IN.^2$ UNCERTAINTY IN BALANCE FOR EACH PROBE	
TOTAL PROBE WEIGHT ALLOCATION -	544 KG (1200 LB)	
MECHANICAL		
LARGE PROBE -	THREE POINT ATTACHMENT AND RELEASE	
SMALL PROBE -	FOUR PAIRS OF PADS APPROXIMATELY 90° APART	
THERMAL -	0 TO 90°F	
SEPARATION ATTITUDE AND DYNAMICS (3)		
	LP	SP
MAXIMUM ANGULAR MOMENTUM VECTOR DIRECTION ERROR	2°	3°
MAXIMUM SPIN AXIS NUTATION ANGLE ABOUT MOMENTUM VECTOR	2°	3°
MAXIMUM SPIN RATE ERROR	± 0.1 RPM	± 0.2 RPM
MAXIMUM VELOCITY ERROR	0.02 M/S	0.05 M/S
UMBILICAL		
POWER:	LP/48 WATTS @ +28 VDC ± 10% SP/13 WATTS @ +28 VDC ± 10%	
COMMANDS:	LP/SERIAL 16 BIT WORD, 1 BIT/S SP/SERIAL 16 BIT WORD, 1 BIT/S	
TELEMETRY	LP/192 BIT BLOC, 128/256/512 BIT/S SP/192 BIT BLOCK, 512/16 BIT/S	

Table 5-1D. Mission Design Requirements Summary

TARGETING	LP: AS CLOSE TO SUBSOLAR POINT AS POSSIBLE NO MORE THAN 70° FROM SUBSOLAR AND NEAR EQUATOR SP: GOOD LATITUDE AND LONGITUDINAL DISPERSAL PROBE ENTRY SITES: WITHIN 55° OF SUBEARTH
LAUNCH/ARRIVAL	LAUNCH WINDOW: AUGUST 20 TO 29, 1978 (5 MINUTES EACH DAY) ARRIVAL: DECEMBER 17, 1978 ATLAS/CENTAUR INJECTED MASS: 790 KG (1742 LB)
INTERPLANETARY CRUISE	119 TO 110 DAYS
PROBE RELEASE SEQUENCE	SEQUENTIAL LP E_1 - 25 DAYS SP1 E_1 - 21 DAYS SP2 E_2 - 17 DAYS SP3 E_2 - 13 DAYS ($E_2 = E_1$ + 90 MINUTES)
PLANETARY APPROACH	INITIAL RADIUS (10^6 KM): LP - 10.7; SP - 9.0 TO 5.6 EARTH ASPECT ANGLE: LP - 138° TO 145°; SP - 125° MINIMUM, 152° MAXIMUM SOLAR ASPECT ANGLE: LP - 40° TO 72°; SP - 15° MINIMUM, 55° MAXIMUM VENUS ASPECT ANGLE: LP - 23° TO 44°; SP - 14° MINIMUM TO 61° MAXIMUM

ENTRY PROFILE FOR TARGET SET A (E ≈ 250 KM)				
LAT/LONG	0°/65°	-45°/135°	0°/165°	-22.5°/110°
V_E	11.34 KM/S	11.34 KM/S	11.34 KM/S	11.34 KM/S
E	-35°	-30°	-56°	-41°
E	0°	0°	0°	0°
B_E (KG/M^2)	86.4	141.4	141.4	141.4
PEAK G'S	330	275	465	368
MAX Q(10^5 N/M^2)	2.8	3.8	6.4	5.1
PARACHUTE/INSTRUMENT DEPLOYMENT				
MACH NUMBER	0.786	1.2	0.65	0.85
DYNAMIC PRESSURE (N/M^2)	1695	4405	3016	3399
SCIENCE DEPLOYMENT ALTITUDE (KM)	70.45	70.1	66.4	68.0

DESCENT PROFILE	
LARGE PROBE	
PARACHUTE PHASE BALLISTIC COEFFICIENT	7.85 KG/M^2
STAGING ALTITUDE	42.9 KM
TIME ON PARACHUTE	39.7 MIN
DESCENT CAPSULE BALLISTIC COEFFICIENT	550. KG/M^2
TOTAL DESCENT TIME	73. MIN
SMALL PROBES	
DESCENT BALLISTIC COEFFICIENT	197.9 KG/M^2
TOTAL DESCENT TIME	65. MIN

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 5-2A. Large Probe Design Requirements Summary

1. MECHANICAL
SPIN RATE
10 RPM @ RELEASE TO ENTRY
5 REV/KM - 20 RPM MAX DESCENT
38° MAXIMUM ENTRY ANGLE/32° MINIMUM ENTRY ANGLE
10° MAXIMUM ENTRY ANGLE OF ATTACK
DECELERATION G MAX = 358
STABILITY
ENTRY: STATICALLY AND DYNAMICALLY STABLE
COMMUNICATIONS: + 15° MAXIMUM OSCILLATION ABOVE 30 KM, +8° FROM 30 KM DIMINISHING TO ZERO AT THE SURFACE (NO GUST CONDITION)
SCIENCE: ZERO LIMIT CYCLE OSCILLATION BEHAVIOR (DESIGN GOAL)
PARACHUTE OPERATION: 70 TO 43 KM. SEPARATE DESCENT CAPSULE AND AEROSHELL ZERO GLIDE BEHAVIOR (DESIGN GOAL)
BALLISTIC COEFFICIENTS:
HYPERSONIC: 86.4 KG/M^2 (0.55 SLUG/FT^2)
PARACHUTE: 7.85 KG/M^2 (0.05 SLUG/FT^2)
DESCENT: 550 KG/M^2 (3.5 SLUGS/FT^2)
EQUIPMENT TEMPERATURE:
INTERNAL: 255 TO 339° K (0 TO 150°F)
EXTERNAL: PARACHUTE 228 TO 407°K (-50 TO 274°F 920°K
MATERIALS: NON-OUTGASSING
PRESSURE
INTERNAL: 101 TO 41.4 KN/M^2 (14.7 TO 6 PSIA)
EXTERNAL: 10^{-4} TORR TO 93 ATMOSPHERES
WEIGHT: 264 KG
2. ELECTRICAL
ELECTRICAL POWER
28 VDC ± 10% — 560 W-HR — SEE POWER PROFILE AND POWER ALLOCATION TABLE IN SECTION 7.8
DATA HANDLING AND COMMAND
DATA CHANNELS: 6 DIGITAL/58 ANALOG/29 BILEVEL
ANALOG-DIGITAL CONVERSIONS: 6/7/10 BIT WORDS
DATA STORAGE: 4300 BITS
DATA OUTPUT:
BI-PHASE MODULATED SUBCARRIER
RATE = 1/2, K = 32 CONVOLUTIONAL CODE
256 SYMBOLS PER SECOND
COMMAND CHANNELS: 50
COMMAND BIT RATE: 1 BIT/S
COMMUNICATIONS
TWO-WAY DOPPLER
256 SYMBOLS PER SECOND
PCM/PSK/PM MODULATED S-BAND
ANTENNA POLARIZATION: RIGHT-HAND CIRCULAR
EMC
MODIFIED MIL-STD APPROACH
3. RELIABILITY 0.9

Table 5-2B. Small Probe Design Requirements Summary

1. MECHANICAL
SPIN RATE: 10 RPM @ RELEASE TO ENTRY
60° MAXIMUM ENTRY ANGLE/25° MINIMUM ENTRY ANGLE
10° MAXIMUM ENTRY ANGLE OF ATTACK
DECELERATION G MAX = 488
STABILITY: ± 15° MAXIMUM OSCILLATION ABOVE 30 KM +8° FROM 30 KM DIMINISHING TO ZERO AT THE SURFACE (NO GUST CONDITION). ZERO LIMIT CYCLE OSCILLATION BEHAVIOR (DESIGN GOAL)
BALLISTIC COEFFICIENT: 141.4 KG/M^2 (0.9 SLUG/FT^2)
EQUIPMENT TEMPERATURE:
INTERNAL: 255 TO 339°K (0 TO 150°F)
EXTERNAL: 920°K
MATERIALS: NON-OUTGASSING
PRESSURE:
INTERNAL: 101 TO 41.4 KN/M^2 (14.7 TO 6 PSIA)
EXTERNAL: 10^{-4} TORR TO 93 ATMOSPHERES
WEIGHT: 70 KG EACH !
2. ELECTRICAL
ELECTRICAL POWER
28 VDC ± 10% — 274 W-HR — SEE POWER PROFILE AND POWER ALLOCATION TABLE IN SECTION 7.8.
DATA HANDLING AND COMMAND
DATA CHANNELS: 6 ANALOG/48 DIGITAL/32 BILEVEL
ANALOG-DIGITAL CONVERSION: 6/7/10 BIT WORDS
DATA STORAGE: 1900 BITS
DATA OUTPUT:
BI-PHASE MODULATED SUBCARRIER
RATE = 1/2, K = 32 CONVOLUTIONAL CODE
128 SYMBOLS PER SECOND
COMMAND CHANNELS: 25
COMMAND BIT RATE: 1 BIT/S
COMMUNICATIONS
ONE-WAY DOPPLER
128 SYMBOLS PER SECOND
PCM/PSK/PM MODULATED S-BAND
ANTENNA POLARIZATION: RIGHT-HAND CIRCULAR
EMC
MODIFIED MIL-STD APPROACH
3. RELIABILITY 0.9

and risk. All probe designs were developed by a synthesis process that combined various options of the following major items to form each configuration: aeroshell shape, descent capsule configuration, decelerator/staging, and electronics complement.

The pre-midterm studies, which led to the selection of preferred Thor/Delta and Atlas/Centaur probe configurations, are highlighted in

Sections 5.1.3 and 5.1.4. These configurations were developed on the basis of the Version III science payload definition and a 1977 probe mission launch date. The final preferred probe configurations, presented in Section 5.1.1, reflect decisions by NASA regarding the science payload complement, launch vehicle, and probe mission launch date, which were made after the midterm review. The results of the pre-midterm studies formed the basis for the final probe configurations. Section 5.1.4.4 summarizes the rationale for this design.

5.1.1 Preferred Atlas/Centaur Probe Configurations

The preferred large and small probe configurations for the specified 1978 Atlas/Centaur launch, carrying the Version IV nominal science payload, and reflecting the final results of the Phase B probe design effort, are illustrated in Figure 5-3. Simplified large and small probe block diagrams indicating the principal electrical functions and interfaces are shown in Figure 5-4A and 5-4B.

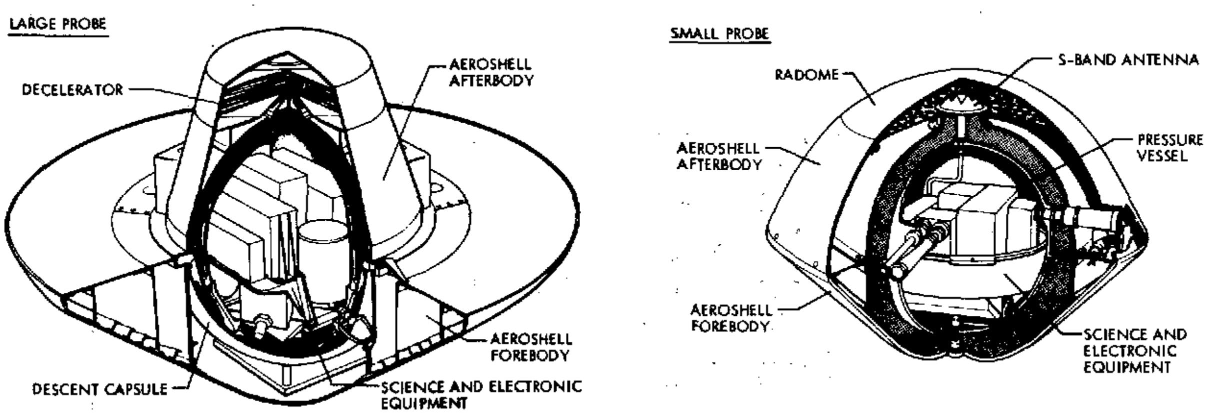


Figure 5-3. Final Preferred Probe Configurations

These configurations incorporate many features which will reduce costs in comparison to probes design for a Thor/Delta launch vehicle. Ways in which lower costs were achieved include the following:

- Increased design margins.
- Maximum utilization of flight proven designs, materials, and hardware.
- Commonality of equipment between large and small probes and between probes and bus/orbiter.

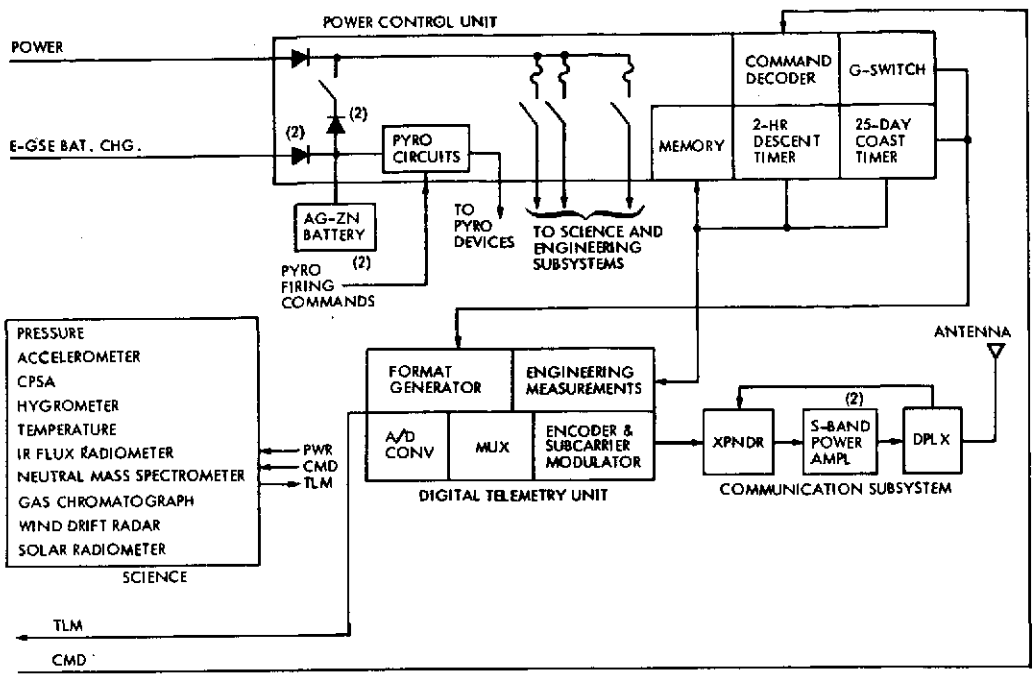


Figure 5-4A. Atlas/Centaur Final Preferred Large Probe Block Diagram

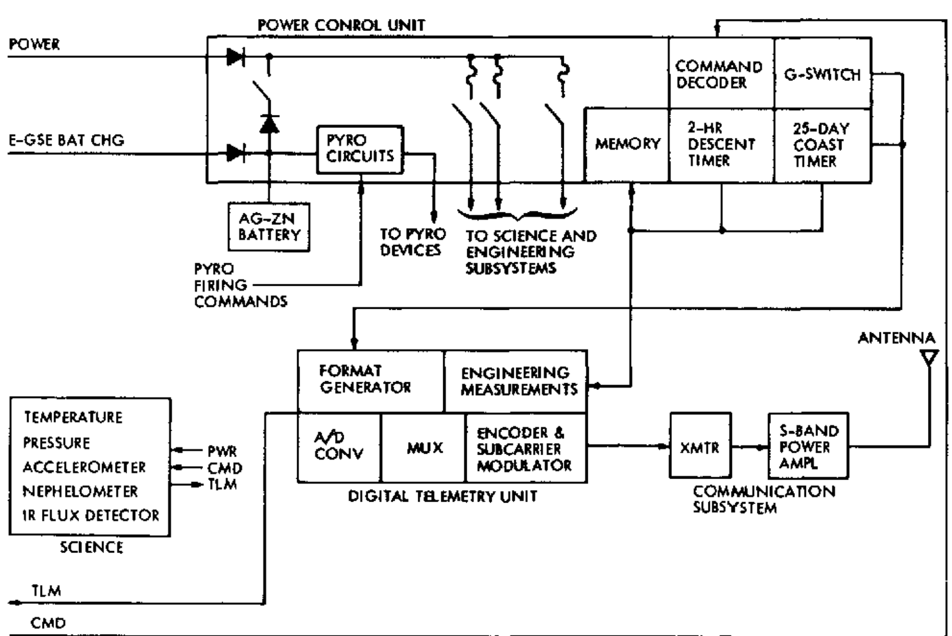


Figure 5-4B. Atlas/Centaur Final Preferred Small Probe Block Diagram

- Easy accessibility to probe instruments and subsystems, to facilitate integration and test.
- Minimizing new developments and test requirements.

These approaches with specific examples are discussed more fully in Section 11. Section 6.1 defines in detail the preferred Atlas/Centaur probes. The configuration options and trade studies that led to this selection and to the preferred design for Thor/Delta probes are presented in the following sections.

5.1.2 Configuration Options

Probe configuration concepts developed during this study resulted from a synthesizing process that combined various aerodynamic shapes, decelerator staging schemes, and stabilization techniques with other selected subsystem design approaches. All probe concepts studied were configured to meet the science data, mission, and system design requirements, as limited by the launch vehicle payload weight constraint. The other main factor driving the design concepts was the development of a low-cost approach with the least impact on risk as well as performance.

5.1.2.1 Entry Configuration

Aeroshell Forebody

Aeroshell forebodies, consisting of blunted large angle cones were the basic shapes studied. Cone half-angles of 1.22, 1.05, 0.96 and 0.79 rad (70, 60, 55 and 45 degrees) were examined. The 1.22 rad (70 degrees) aeroshell forebody was included since considerable aerodynamic data exist for this shape from the Viking program. Considerable data also exist for 1.05 rad (60 degrees) cone shapes both from the Viking test program and other sources, and 0.96 rad (55 degrees) cone data are available from PAET. To optimize the reflective heat shield performance and thus try to reduce heat shield weight, a spherical segment (Apollo type) forebody was also examined.

Aeroshell Afterbody

The aeroshell afterbody configurations are based on considerations of aerodynamic stability, weight, equipment integration, and interface with

the probe bus. A hemispherical afterbody is a simple stable shape. However, tailored afterbodies afford advantages in the area of lower weight and improved low-speed limit cycle behavior, as well as facilitate attachment to the probe bus.

Existing versus New Configurations

The PAET, Viking, and Apollo aeroshapes were examined in this study with the intent of making use of their available aerodynamic data, thereby reducing development testing costs. These were traded against new configurations tailored specifically to the probe mission requirements.

Identical versus Distinct Large and Small Probe Shapes

Common large and small probe shapes were traded against individual shapes for each. Factors entering into the tradeoff are development testing required, design change costs, entry heating, internal equipment integration, probe bus installation, aerodynamic flight characteristics, and staging techniques.

5.1.2.2 Descent Capsule

Spherical versus Nonspherical Pressure Vessels

A sphere is a simple, easily manufactured, and structurally efficient pressure vessel. Nonspherical pressure vessels were also considered in the attempt to maximize pressure vessel design commonality for the large and small probes and to facilitate obtaining desired c.g. locations.

Internal versus External Thermal Insulation

A weight comparison was made between a heavier, denser external thermal insulation and a lighter, less dense material internal to the pressure vessel. Consideration of the pressure shell temperature, internal pressure, and science integration problems were the other factors involved in the tradeoff.

Pressure versus Nonpressure Protected Approach

A variety of pressure and nonpressure protected concepts were evaluated. Pressure equalized (i.e., nonpressure protected) designs included concepts with filler materials, with atmospheric venting and phase change

heat exchangers, and with cold gas pressurization. A pressure vessel preloaded to approximately half the Venus pressure and several completely pressure protected designs were compared against the pressure equalized designs.

Fin versus Ring Stabilization (Large Probe)

Fin stabilized descent capsule configurations were traded off against ring stabilized versions. Fin number, length, and deflection angle was varied to obtain optimum aerodynamic characteristics, evolving finally into a perforated flare configuration. Ring diameter and height above the sphere major diameter were the aerodynamic variables for this class of configurations. The two approaches were traded on the basis of aerodynamic performance, impact on aeroshell configuration, and science/subsystem integration complexity.

5.1.2.3 Staging and Decelerator Options

Staged versus Unstaged, and Staging Techniques

Two versions not requiring parachutes were evaluated for the large probe. A mechanically staged concept in which the descent capsule is released to expose the science instruments was compared to an unstaged concept in which the aeroshell was retained and the science deployed through the aeroshell or ports jettisoned to expose the instruments. These no-parachute concepts were then evaluated against the parachute designs for risk, performance, and cost.

One- versus Two-Stage Decelerator

A supersonic drogue stage was evaluated as a means to increase the altitude at main chute deployment and augment the basic probe stability. This was compared to a single stage subsonic main chute.

Canopy Types

Disk-gap-band, ringslot, ringsail, ribless guide surface, ribbon, and cross-type parachute canopies were considered. Factors involved in the selection included drag/weight efficiency, stability, lift (glide) characteristics, ballistic coefficient, opening load characteristics, transonic speed operation and test history.

Existing Designs

Parachutes from the Discoverer, Biosatellite, Viking, and Apollo programs, plus existing aircraft decelerator, cargo, and man-carrying designs were surveyed for applicability. A retro system was also studied.

Stowage/Deployment

A variety of schemes were derived to stow and deploy the parachutes. All were complicated by the goal to place both the communication antenna and the parachute on the centerline. Versions were evaluated employing parachute canisters located at various positions in the forebody and afterbody area. Deployment methods studied included pilot chute, mortar, drogue slug gun, gas ejection, catapulting, and rocket extraction.

5.1.2.4 Electronics Complement

New versus Existing Hardware

Weight, volume, and cost were the key considerations in evaluating new designs against existing hardware to perform communication, data handling and command and power distribution functions in both probes. Sources of applicable hardware were Viking, Pioneers 10 and 11, and classified programs. Other factors considered were degree of modification required to sustain entry decelerations and applicability to both large and small probes.

Common versus Custom Hardware

Commonality of electronic hardware between the large and small probes and the probe bus/orbiter was evaluated with respect to custom designs. Cost, weight, and size were the primary evaluation factors.

Subsystem Options

The tradeoffs itemized below and performed at the subsystem level were not major configuration selection drivers, but did impact electrical system definition and interfaces. These lower level trades and analyses are covered in detail in Section 7.

- Regulated versus unregulated power bus
- Monovalent versus divalent battery charging

- Centralized versus decentralized programmer
- Centralized versus decentralized analog-to-digital converter
- MFSK versus PSK/PM modulation
- Convolutional versus Viterbi coding
- One-way versus two-way Doppler tracking

5.1.3 Thor/Delta Tradeoffs

Twelve configurations for the large probe and three for the small probes were examined and have been fully reported in MMC Technical Note P73-203434-053, "Probe Configuration Study Summary" submitted earlier to NASA/ARC. This technical note is highlighted in this and the following section.

The major options considered for the large probe were various decelerator systems, aeroshell shapes, and descent capsule stabilization techniques as described in Section 5.1.2. These probe configurations are shown in Figure 5-5 (A through E). The Phase B proposal configuration featured a 2.1-meter pilot parachute and a 7.6-meter main parachute located in a toroidal canister. This configuration was then updated with the science payload defined in the Phase B study specification to provide a baseline design. This primarily required a transmitter power increase from 10.9 to 20 watts to accommodate the higher science sampling rates. Seven of the configurations utilized a 2.1-meter pilot parachute with a 3.5-meter main parachute located in a smaller toroidal canister. The 3.5-meter parachute is the minimum size required to allow the descent capsule to be extracted from the aeroshell with a 1-g differential acceleration. In addition, the pilot and main parachute locations, aeroshell shape, and descent capsule configuration were varied. Two no-parachute concepts were also studied. One featured a mechanically separated aeroshell to allow for descent capsule release through the aeroshell, while the other configuration retained the aeroshell and exposed individual science instruments via sampling ports. Another configuration compared pilot deployment versus direct mortaring of the main parachute, i.e., the 3.5-meter parachute is used with and without the 2.1-meter pilot parachute. The 3.5-meter parachute can either be located in a small cylindrical canister or in a mortar canister.

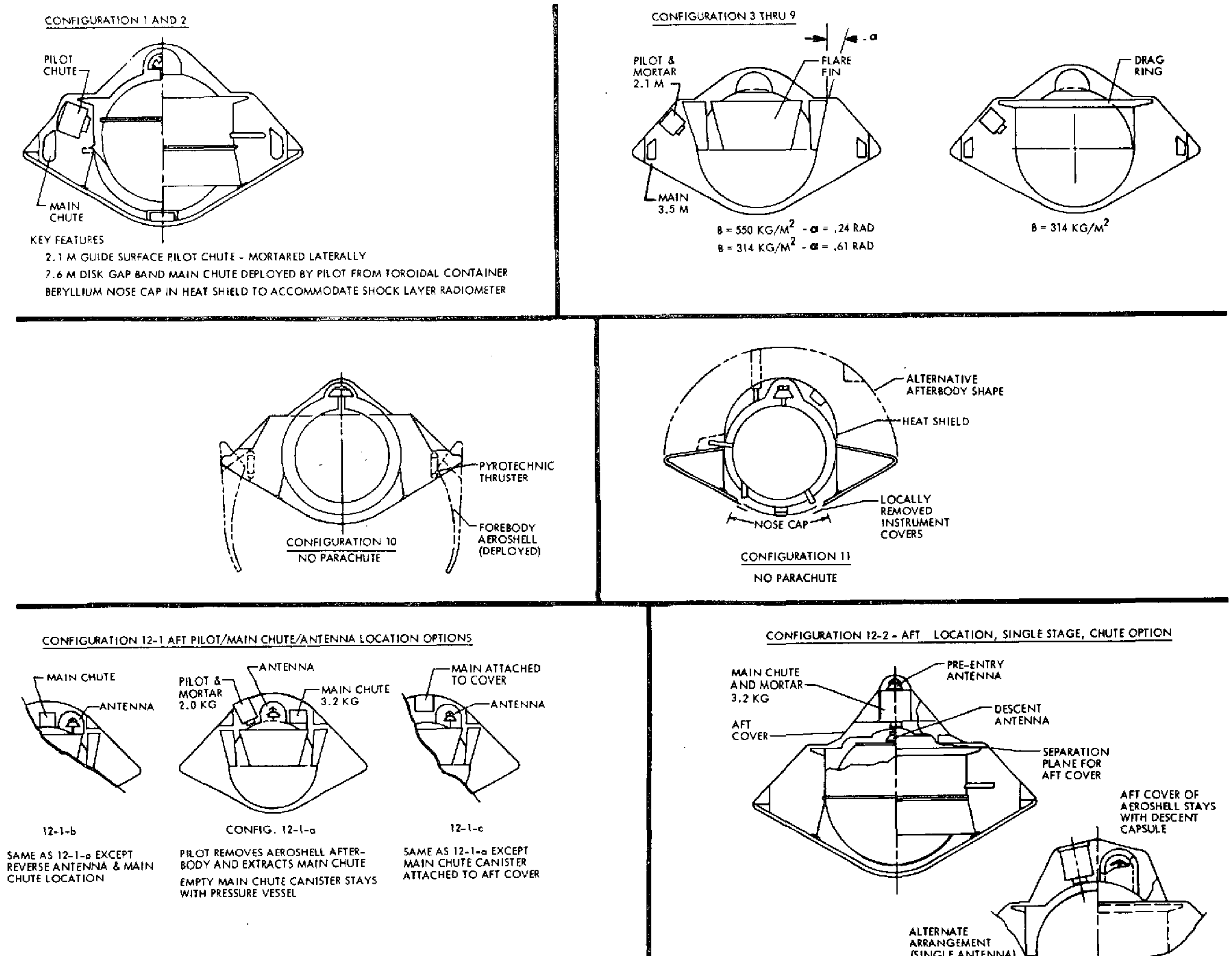


Figure 5-5. Thor/Delta Large Probe Configurations

The major configuration variable considered for the small probe is the aeroshell shape. Three small probe configurations were studied (Figure 5-6): one is an updated Phase B proposal configuration; the second has a PAET afterbody shape; and the third has a 1.05 rad (60 degrees) half-angle forebody and conical/spherical cap afterbody.

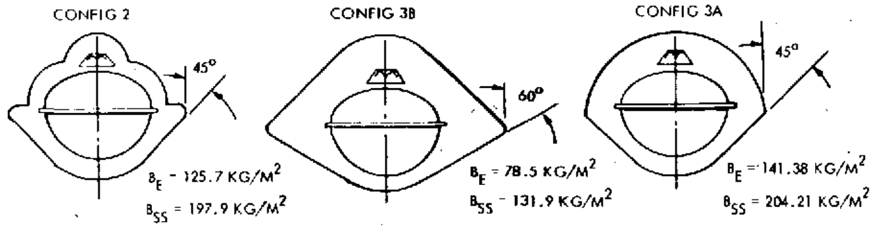


Figure 5-6. Thor/Delta Small Probe Configurations

5.1.3.1 Results of Thor/Delta Configuration Studies

The configuration study results are summarized by the configuration options defined in Section 5.1.2; a more complete description of the trade-off studies can be found in the MMC technical note mentioned previously.

Aeroshapes

The large probe forebody shapes evaluated were 0.96, 1.05, and 1.22 rad (55, 60 and 70 degrees) half-angle blunted cones; and a spherically blunted shape similar to the Apollo entry vehicle. The 1.22 rad (70 degree) half-angle cone has so little volume in the forebody that maintaining an acceptable c.g. location for stability purposes requires too large and heavy an aeroshell. It also has a much higher radiative heating input. The Apollo shape was discarded due to stability and packaging problems, as well as the high radiative heating. The 1.05 rad (60 degree) cone shape, for which considerable aerodynamic data exist, provided adequate volume, acceptable heating input, and good entry stability, and had slightly greater drag than the 0.96 rad (55 degrees) shape. Thus, the 1.05 rad (60 degrees) half-angle cone was selected as the preferred forebody shape.

The small probe forebody shapes evaluated were the 0.79 rad (45 degrees) half-angle and the 1.05 rad (60 degrees) half-angle common to the large probe. The 1.05 rad (60 degrees) half-angle aeroshell was found to be 5.4 kg heavier than the 0.79 rad (45 degrees) half-angle because of the greater descent time and the larger diameter required to meet the c.g.

location requirements. It was therefore discarded in favor of the 0.79 rad (45 degrees) shape.

The afterbody shapes considered were the hemispherical (PAET) shape, cone/sphere segment shapes, and biconic shapes. The PAET afterbody was not selected for the large probe due to the increased weight and complexity aspects of integrating the large spherical shape with the spacecraft bus. The selected smaller 0.79 rad (45 degrees) cone afterbody shape provided sufficient volume to house the selected flare-stabilized version of the descent capsule and exhibited good entry stability characteristics. In the case of the small probe, the 0.52 rad (30 degrees) cone/spherical cap afterbody was selected over the PAET type because it provided better low-speed oscillation characteristics and adequate pitch damping characteristics.

Decelerator Types

Two configurations with the 7.6-meter main parachute stowed in an equatorial toroidal canister were discarded because the system was 7.5 kg heavier than the minimum parachute required for separation and descent velocity control. Also, deploying the parachute from the toroidal canister was estimated to require a more costly development. Consequently all configurations involving the toroidal canister were dropped. With the small main chute (3.5-meter diameter) selected, the near aft centerline chute locations became feasible and desirable. A configuration employing a pilot chute was developed when the aeroshell afterbody had to be removed to allow for an up-looking solar radiometer window and the aureole/extinction detector was mounted on top of the descent capsule and rotated internally. With the subsequently redefined solar radiometer as a solar flux radiometer that could utilize a horizontal-looking window, and the decision to spin the descent capsule instead of rotating the aureole/extinction detector, thereby allowing a horizontal-looking window, it was found that the afterbody could be retained during parachute descent. For this science arrangement a decelerator system consisting of main parachute mortared directly becomes the preferred design. For this size parachute (3.5-meter diameter) this approach is the lightest, least costly, and most reliable system. In it the afterbody remains attached to the descent capsule until the parachute is staged.

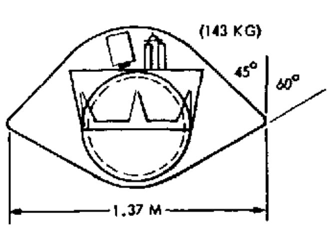
Two no-parachute systems were evaluated and discarded. The rationale for this decision is: increased weight (7.7 to 11.8 kg), costs, and risks associated with mechanically separating the aeroshell; and the inadequate science sampling capability of the unstaged configuration, which could cause inaccuracies because of converging channeled flow, contamination from heat shield, and residual atmospheric samples retained in the science ports.

Descent Capsule Stabilization

The tradeoff study between the thin rings and the final flares was based on spin tunnel and other aerodynamic test results. The flares provided more flexibility in ballistic coefficient control and greater resistance to tumbling. Subsequent to the Thor/Delta portion of the design study, the use of a centrally located perforated ring of finite thickness was found to be a simpler, more effective design than either the thin rings or the flares.

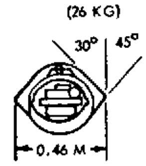
5.1.3.2 Recommended Thor/Delta Probe Configuration Summary

The probe configurations recommended for the Thor/Delta launch vehicle are a large probe configuration with a single stage deceleration and a small probe configuration with a modified (cone/spherical cap) afterbody. The configuration drawings depicting this design are shown in Figure 5-7 and 5-8. The functional block diagrams are shown in Figures 5-9 and 5-10. The selected configurations meet the mission and system design requirements with the lightest weight, least cost, and most reliable design of the systems evaluated.



KEY FEATURES
BALLISTIC COEFFICIENT (KG/M²):
ENTRY 70.7
PARACHUTE 18.9
DESCENT CAPSULE 550
PARACHUTE: SINGLE STAGE, 3.51 M DIAMETER
PRESSURE VESSEL: 0.59 M ID ALUMINUM SPHERE
BATTERY: 207 W-HR
TRANSMITTER: 20 WATTS MINIMUM

Figure 5-7. Thor/Delta Large Probe



BALLISTIC COEFFICIENT (KG/M²):
ENTRY 125.6
TERMINAL DESCENT 204.2
PRESSURE VESSEL: 0.229 M X 0.267 M TITANIUM OBLATE SPHEROID
BATTERY: 83 W-HR
TRANSMITTER: 9.1 WATTS MINIMUM

Figure 5-8. Thor/Delta Small Probe

5.1.4 Atlas/Centaur Tradeoffs

The primary objective of using the increased capabilities of the Atlas/Centaur launch vehicle for probe configuration studies was to emphasize low cost. Ground rules precluded use of the added weight capability to

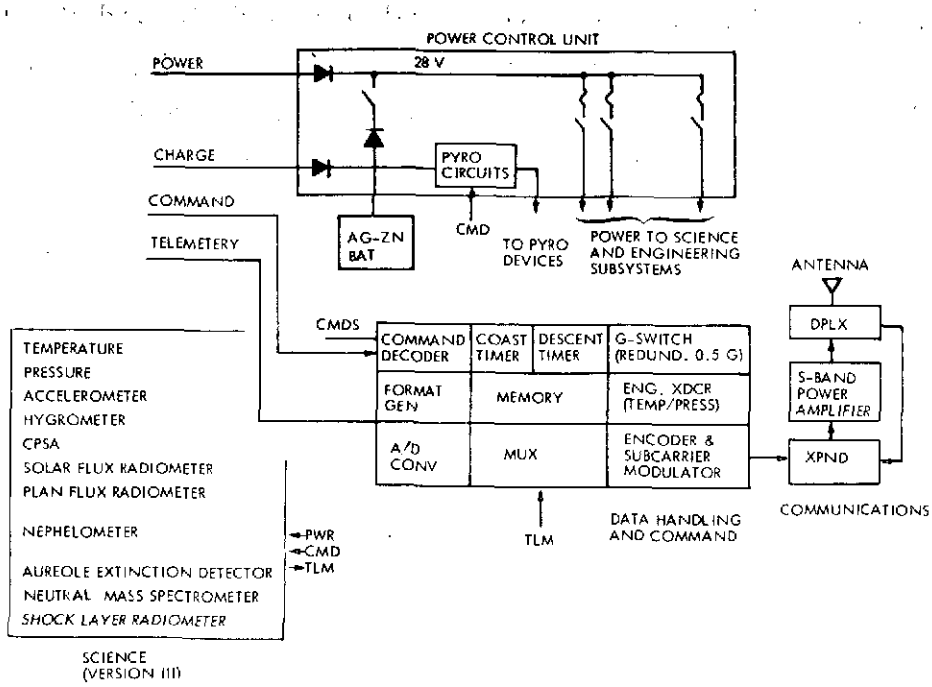


Figure 5-9. Recommended Thor/Delta Large Probe Block Diagram

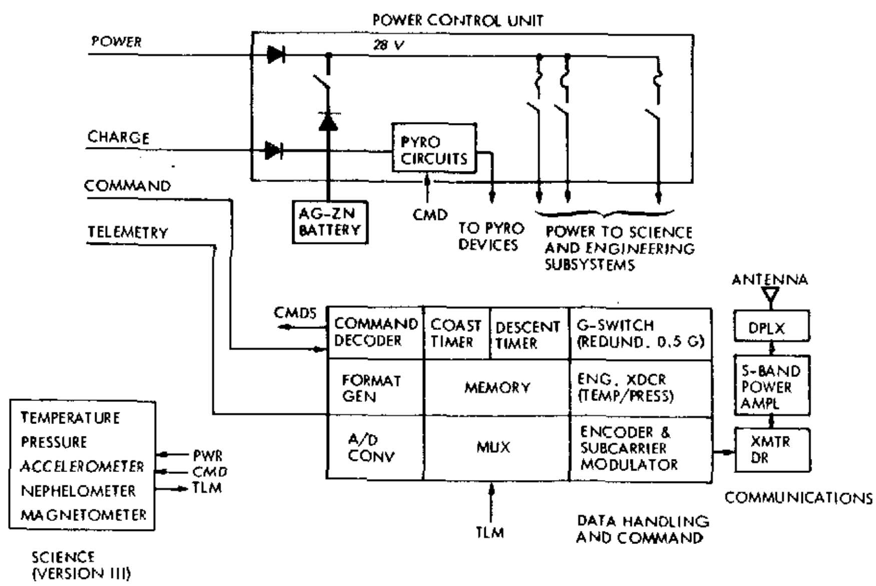


Figure 5-10. Recommended Thor/Delta Small Probe Block Diagram

enhance or modify the mission or science accomplishments of the system defined for the Thor/Delta launch vehicle. The approach was to use the increased weight and volume available to increase design margins, to use existing designs to a maximum, and to reduce the design, development, and test efforts to arrive at the lowest possible cost. The discussion that

follows highlights the studies conducted up to the midterm review (documented more fully in MMC Technical Note P73-203434-053, "Probe Configuration Study Summary"), and reflects the Version III science payload. The final evolution of the preferred Atlas/Centaur probe configurations from the midterm designs is then traced in Section 5.1.4.4.

5.1.4.1 Candidate Configurations

Nine major probe configurations were examined with minor tradeoffs performed on the subsystems to develop the lowest program cost. The system variations resulted from synthesizing configurations by combining various aerodynamic shapes and decelerator concepts with selected subsystem design approaches.

The first configuration developed was essentially an expanded Thor/Delta configuration. Increased size was used to accommodate existing electronic packages, with higher safety and design factors to reduce design, development, and testing. The next two configurations used various existing aeroshapes and parachutes. Aeroshapes that were considered included Viking, Apollo, and PAET. Existing parachutes evaluated included those from Viking, Apollo, aircraft cargo, aircraft decelerators, and satellite recovery. With these three configurations, the lowest cost approach was sought by optimizing the electronic hardware packages. The next block of configuration tradeoffs evaluated a two-stage versus single stage parachute versus an unstaged approach. In an attempt to reduce peak entry deceleratic levels and thereby provide some weight and volume capability to accommodat more existing electronics hardware, two additional configurations were examined. One used retrothrusters and the second employed a lifting aerodynamic shape. The final configuration concept used the added weight and volume capability of the Atlas/Centaur launch vehicle to accommodate an alternative Version III science payload, which made maximum use of existing instruments to minimize the science development costs.

5.1.4.2 Results of Configuration Studies

Aeroshapes

The PAET forebody shape was adopted without change for both large and small probes. However, the mounting of the probes into the bus

required tailoring the afterbody to facilitate installation into the bus. The PAET afterbody shape is hemispherical, but was modified to a conic shape with a spherical cap around the antenna. All other shapes were discarded. The Viking configuration placed the c. g. too far aft for a stable design. The Apollo version is not stable at subsonic speeds. The lifting entry analysis indicated that the maximum peak load reduction is on the order of 15 percent. This approach was not considered practical in view of the added complexity of the required attitude control system. The reduction of entry velocity by a retro-maneuver included schemes with a bus maneuver prior to probe release, probe release from an intermediate orbit, and a probe retro-maneuver at entry. None of these schemes provided enough peak load relief, considering the fuel weight penalty. At best the peak load was reduced to a 146-g level, with a fuel weight penalty equal to the weight of the probes. The mechanically staged or unstaged versions were discarded for the same reasons as in the Thor/Delta study.

Decelerator Types

Several combinations of parachute options were evaluated for the large probe. The Viking 15.24-meter disc gap band parachute was a misapplication for a direct mortar single stage deployment, because it is unable to withstand the 1436.4 N/m^2 dynamic pressure at inflation. It is designed for a maximum dynamic pressure of 526.7 N/m^2 and is unstable at descent velocities below 3.05 m/s. The descent velocity of a fully inflated Viking parachute at 6090 km would be 2.44 m/s.

Each Apollo main parachute is a 26.06-meter, 58.97 kg ringsail type and the pilot parachute is a 2.19-meter, 11.34 kg ringslot parachute. The weight of the Apollo main parachute obviated its use. However, the Apollo drogue parachute combined with a Viking parachute was a potential candidate for a two-stage deployment scheme. This configuration was acceptable, but was heavy and resulted in a slower descent than desired. The Apollo drogue, with a 4.42-meter diameter ringslot was the most viable design for a two-stage deployment approach. The best weight and performance was achieved using a new 2.44-meter pilot and a new 4.27-meter main parachute design; however, cost consideration eliminated this version. The no-parachute design was also dropped based on the Thor/Delta

conclusion. The single stage 4.42-meter diameter ringslot parachute was selected as the preferred configuration (existing aircraft deceleration parachute).

Descent Capsules

To permit relaxing tight fabrication tolerances and eliminate the structural test model, the safety factor was increased from 1.0 to 1.25 times the limit load impressed at 9.49 MN/m^2 (93.6 atmospheres). The Thor/Delta approach was to design for failure at 9.49 MN/m^2 (93.6 atmospheres), the pressure at the Venus mean surface level.

Three pressure vessel configurations were evaluated for lowest cost. Individually sized spheres to accommodate the large and small probe payloads were the lightest and smallest. Using a common end dome with a length of cylindrical center section sufficient to enclose the required probe volume was heavier, but permitted easier access for installation. The third design, the Thor/Delta configuration, was selected based on ease of installation and access. The cost of all three was essentially the same.

Heat Shield/Thermal Control

Increasing the heat shield safety factor from 1.4 to 1.6 and using a more effective, but denser, heat shield material would reduce the development and material costs 15 to 20 percent and eliminate all of the wind tunnel heat transfer tests. Thermal insulation thickness was increased to reduce the shell temperature from 625 to 562^{o}K for a safety factor of 1.37. This provides a cost reduction in the thermal balance tests for the post-separation, cruise, and descent phases.

Electrical/Electronics

Commonality of battery design was achieved by satisfying the small probe requirements with one battery design and the large probe requirements with two of these connected in parallel. To take advantage of a low-cost "quantity buy," the probe bus will also use the same battery cell, though not the same battery design.

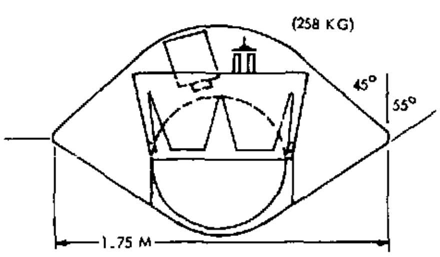
Data handling system designs existing from IMP, MVM '73, Viking, OSO, and Pioneers 10 and 11 were surveyed. The Pioneer hardware was

selected because of lower cost, weight, and power, better magnetic properties, and performance factors. The same unit would be used in both probes.

Candidate transponders were evaluated from the Pioneers 10 and 11, Mariner, and Viking. The Viking unit was selected as most compatible with the least modification required for qualification, and most up-to-date design approach. Companies prominent in the design of power amplifiers for space applications were surveyed, but no existing qualified units were found to meet the probe requirements. One existing unit would be qualified for the small probe and two identical units used in parallel for the large probe. This approach will achieve hardware commonality and meet performance requirements.

5.1.4.3 Recommended Pre-Midterm Atlas/Centaur Probe Configuration

The PAET forebody with modified afterbody was selected as the common aeroshell configuration for both large and small probes, thereby reducing the amount of wind tunnel testing required. The selected single-stage parachute configuration is the preferred design and meets the mission and system design requirements with the lightest weight, least cost, and most reliable design. The configuration drawings depicting this design are shown in Figures 5-11 and 5-12 for the large and small probes. The selected design has achieved the lowest cost by applying the increased weight and volume capability of the Atlas/Centaur launch vehicle to gain increased safety factors and design margins. Common electronics and "existing design" components are used for both probes. The functional block diagrams are shown in Figures 5-13 and 5-14.



KEY FEATURES

BALLISTIC COEFFICIENT (KG/M^2)
ENTRY 86.4
PARACHUTE 18.8
DESCENT CAPSULE 550

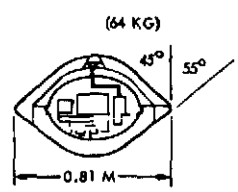
PARACHUTE: SINGLE STAGE 4.42 M AIRCRAFT DECELERATOR

PRESSURE VESSEL: 0.61 MM ID ALUMINUM SPHERE

BATTERY: 286 W-HR

TRANSMITTER: PARALLELED 20 WATT POWER AMPLIFIERS (36 W TOTAL)

Figure 5-11. Recommended (Midterm) Atlas/Centaur Large Probe Configuration



BALLISTIC COEFFICIENT (KG/M^2)
ENTRY 113.9
TERMINAL DESCENT 188.5

PRESSURE VESSEL: 0.35 M X 0.47 M ALUMINUM OBLATE SPHEROID

BATTERY: 132 W-HR

TRANSMITTER: 20 WATT POWER AMPLIFIER

Figure 5-12. Recommended (Midterm) Atlas/Centaur Small Probe Configuration

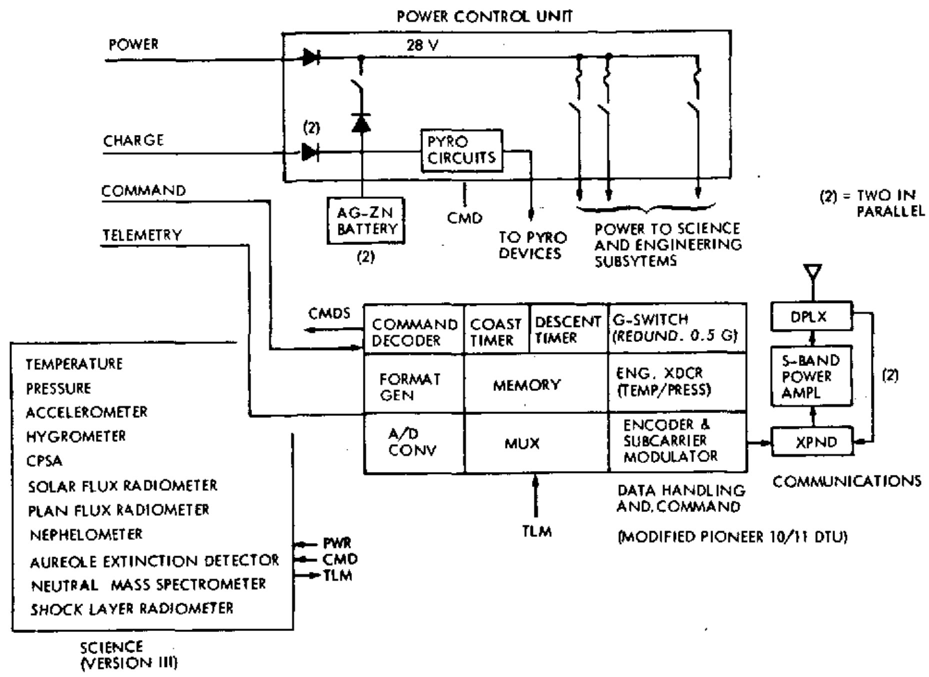


Figure 5-13. Recommended (Pre-midterm) Atlas/Centaur Large Probe Block Diagram

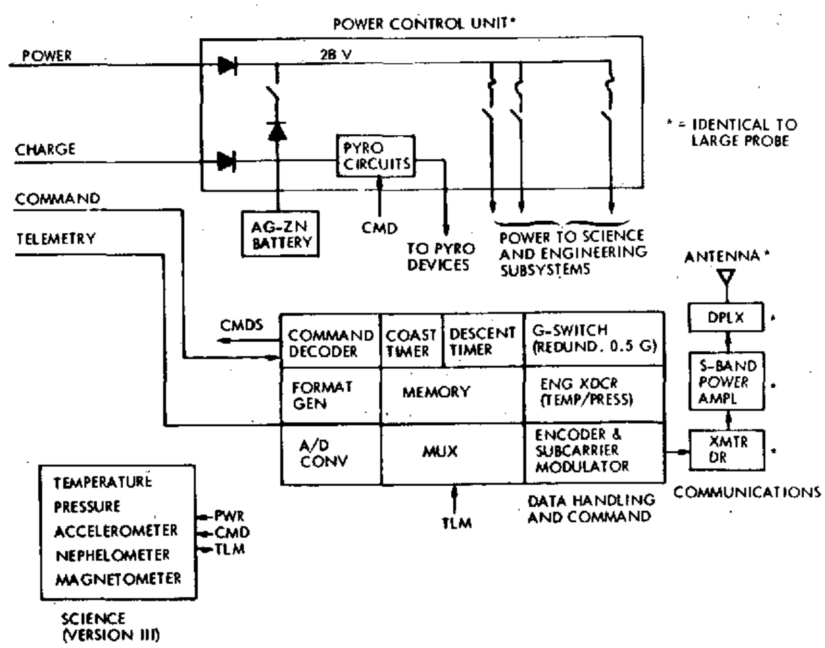


Figure 5-14. Recommended (Pre-midterm) Atlas/Centaur Small Probe Block Diagram

The use of a science payload consisting of a mix of existing and new instruments, plus common instruments where possible, for both probes appears promising as a cost reduction approach. The recommended

(pre-midterm) Atlas/Centaur configuration allows incorporating existing science instruments applicable to the probe mission.

5.1.4.4 Development of the Final Atlas/Centaur Preferred Configuration - Evolution from Midterm Design

Many of the conclusions from the Thor/Delta tradeoff studies and the pre-midterm Atlas/Centaur study activities are embodied in the final preferred Atlas/Centaur design. However, evolution from the midterm design to the final design was influenced by several additional factors and tradeoffs. These contributed to the selection of a final design, which in some areas is quite different from that identified at the midterm review. The primary factors involved were:

1) The definition of Version IV science payload.

2) Further memory-complexity versus descent time tradeoffs.

3) Additional descent capsule aeroshape development test data.

4) The higher entry velocities of the 1978 launch opportunity.

5) Further evaluation of parachute canopy design relative to the wind velocity measurement experiment.

Evaluation of the first two items resulted in a decision to employ a slower descent during the parachute phase to simplify the memory and data system design, and led to the definition of a considerably larger (6.6-meter diameter) parachute. This, in turn, caused a re-examination of the parachute stowage and deployment concepts as they influenced afterbody and antenna configuration. Independent of this parachute/antenna configuration activity, it was determined that a better solution to descent capsule aerodynamic stabilization was afforded by the simple central perforated disk device than by the previously selected flare/fin arrangement. This opened up the possibility for a more compact aeroshell afterbody design that improved aeroshell staging reliability and facilitated closer coupling of probe to bus. These possibilities were evaluated in conjunction with stowing and deploying the larger parachute.

As a result of this study, a sharply cut back afterbody design was selected for the large probe with an aft-centerline located, pilot-deployed

parachute system. This afterbody design was not, however, appropriate to the small probe configuration. The benefits of aerodynamic commonality between probes were, therefore, reviewed, and the conclusion was reached that the cost and performance benefits of the new large probe configuration outweighed those of large and small probe geometric similarity. Other factors in this decision were the opportunity it afforded to revert back to the small probe Thor/Delta shape which performed best in subsonic wind tunnel tests and the likely possibility that forcing identical geometrical shapes on two probes with such distinctly different requirements could lead to problems in the final design period. Other aspects of the evolution to the final design are summarized in Figure 5-15, with more details of the tradeoffs discussed in Section 7.0.

Aside from the overall parachute/antenna/afterbody design considerations described above, additional canopy design tradeoff studies were also performed. These studies stemmed both from the fact that the existing F-105 aircraft decelerator chute previously selected was not compatible with the new, slower descent profile and also from a concern about the wind measurement aspects of the mission. A goal of very low lift/glide behavior during the parachute phase was established in the interests of obtaining valid wind measurement data. A review of canopy designs with this criterion in mind resulted in the recommendation that a ribless guide surface type canopy be used. Finally, no existing chutes were found that were of the right general size range, regardless of canopy type.

5.1.5 Probe Environments

Environmental studies were carried out to determine the best estimate of the environments that the probes are expected to encounter during the mission in terms of maximum ranges or maximum levels. Prior TRW/MMC experience has indicated a need for uniform environmental requirements for the design and verification of space vehicles, plus a need for a single project source for environmental data. With this approach, the probes and all associated equipment are designed to satisfy uniform project environmental design and verification requirements.

The environments were determined by mission phase; i.e., fabrication, assembly and checkout, transportation and storage, prelaunch, launch and trans-Venus injection, interplanetary cruise, Venus encounter, preseparation, separation and probe cruise, Venus entry, and descent. In addition to categorization by mission phase, the expected levels, design limits, and ultimate factors of safety and margins were determined from which environmental design and verification requirements could be generated for the probe components, subsystems, systems, scientific instruments, and scientific experiments.

The Atlas/Centaur Mission Planners Guide, the General Environmental Test Specification for Spacecraft and Components, the Delta Spacecraft Design Restraints, Aerospace Systems Shock Data, and bus/probe environmental interfaces were considered during the generation of information pertaining to the Atlas/Centaur and Thor/Delta launch and interplanetary cruise phase environments. The Models of Venus Atmosphere (1972) was used to generate information pertaining to the near-Venus and Venus environments.

5.1.5.1 Atlas/Centaur Environments

The Atlas/Centaur probes and associated equipment environments were determined in the manner cited above. Table 5-3 summarizes the most significant mission and verification requirements determined during the study. These values along with other environments of interest became the governing values for design personnel throughout the study.

5.1.5.2 Thor/Delta Environments

The Thor/Delta probe and associated equipment environments were determined in the same manner as cited above. Had the Thor/Delta been updated for Version IV nominal probe payloads, the environmental levels would be basically like the Atlas/Centaur levels. The major environmental differences are the lower structural, heat shield and thermal factors of safety for Thor/Delta due to weight criticality, and lower launch vehicle sinusoidal/random vibration levels.

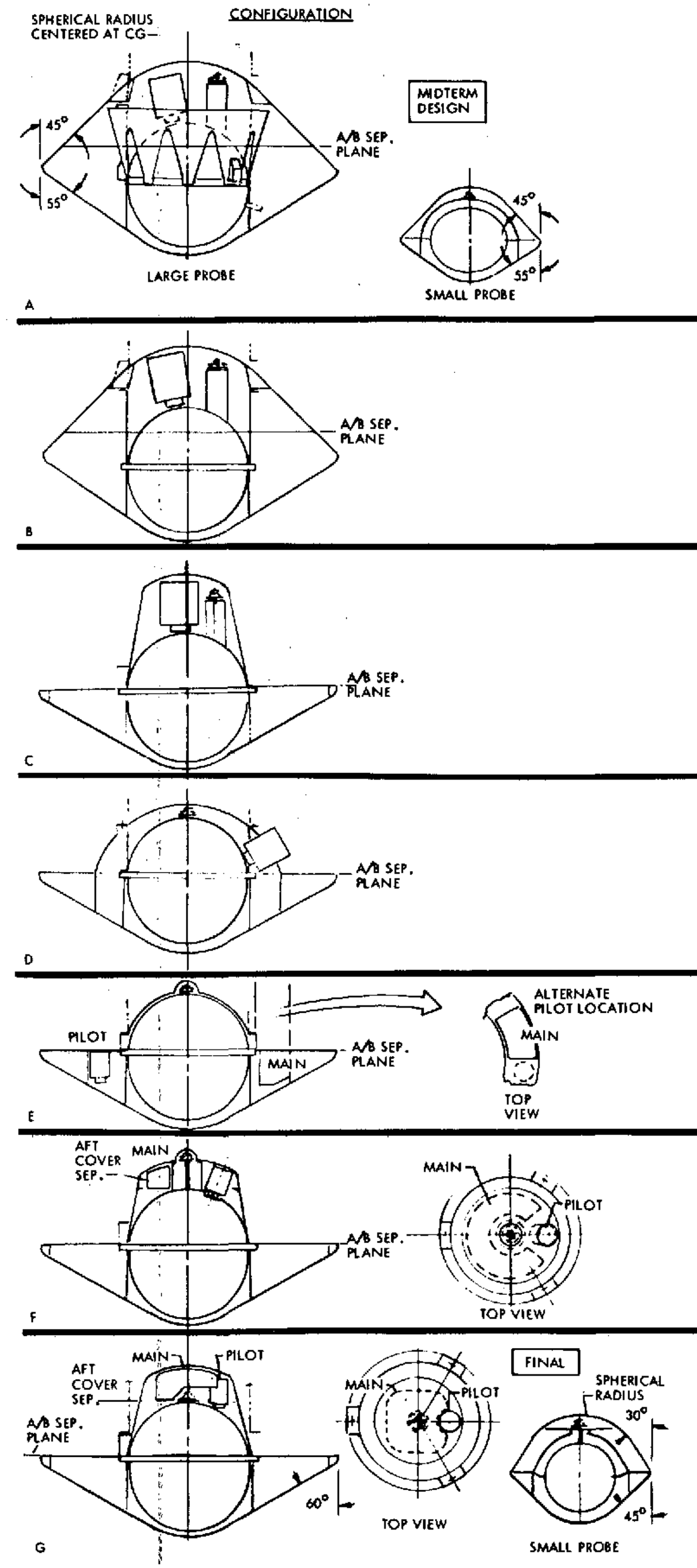
Table 5-3. Atlas/Centaur Probes, Environmental Design and Verification Levels

PROBE ENVIRONMENTS	DESIGN LIMITS		ULTIMATE FACTOR OF SAFETY/MARGIN	
	SMALL PROBE	LARGE PROBE	STRUCTURE (TIMES FLIGHT LEVEL)	PAYLOAD EQUIPMENT (TIMES FLIGHT LEVEL)
DECELERATION				
AXIAL (a)	488 G	358 G	1.56	1.25
LATERAL (a)	8.5 G	2 G	1.56	1.25
ACCELERATION				
AXIAL (b)	20 G		1.56	1.25
LATERAL (b)	12 G		1.56	1.25
SPIN RATE				
LAUNCH (b)	90 RPM		1.56	1.25
PROBE CRUISE (a)	10 RPM		1.56	1.25
TEMPERATURE				
EXTERIOR TO PRESSURE VESSEL				
AFTERBODY (a)	209.8 TO 755.4°K (-82 TO 900°F)	177.6 TO 422.0°K (-140 TO 300°F)	1.0	1.0
FOREBODY (a)	224.3 TO 755°K (-56 TO 900°F)	265.4 TO 541.5°K (18 TO 515°F)	1.0	1.0
PRESSURE SHELL (a)	275.9 TO 403.7°K (37 TO 267°F)	267.0 TO 369.8°K (21 TO 206°F)	1.0	1.0
PRESSURE SHELL EQUIPMENT (a)	255.4 TO 338.7°K (0 TO 150°F)	255.4 TO 338.7°K (0 TO 150°F)		219.3 TO 348.7°K (-65 TO 168°F) PLATE TESTS
PRESSURE				
LAUNCH - PROBE (b)	1.01 N/M² (760 X 10⁻⁵ TORR)		0.001 N/M² (10⁻⁵ TORR)	10.13 MN/M² TO 5.06 MN/M² (1 TO 0.5 ATMOSPHERES)
CRUISE - PROBE (b)	1.33 X 10⁻¹² N/M² (10⁻¹⁴ TORR)		0.001 N/M² (10⁻⁵ TORR)	10.13 MN/M² TO 5.06 MN/M² (1 TO 0.5 ATMOSPHERES)
ENTRY - DESCENT CAPSULE				
EXTERNAL (a)	0.01 MN/M² TO 9.42 MN/M² (0.1 TO 93 ATMOSPHERES)		1175.08 MN/M² (116 ATMOSPHERES)	10.13 MN/M² TO 5.06 MN/M² (1 TO 0.5 ATMOSPHERES)
INTERNAL (a)	10.13 MN/M² (1 ATMOSPHERE)		60.78 MN/M² (6 ATMOSPHERES)	10.13 MN/M² (1 ATMOSPHERE)
VIBRATION				
SINE (b)	AXIAL 1.5 G; 5-15 HZ 4.5 G; 15-21 HZ 1.5 G; 21-100 HZ LATERAL 1.5 G; 5-14 HZ 1.0 G; 14-100 HZ		AXIAL 4.6 G; 5-15 HZ 12.6 G; 15-21 HZ 4.6 G; 21-100 HZ LATERAL 4.6 G; 5-14 HZ 3.0 G; 14-180 HZ 4 MIN/AXIS DURATION	AXIAL 5 G; 5-15 HZ 15 G; 15-21 HZ 7.5 G; 21-35 HZ 5.0 G; 35-50 HZ 3.0 G; 50-100 HZ LATERAL 5 G; 5-30 HZ 3 G; 30-100 HZ 1 MIN/AXIS DURATION
RANDOM (a)	6.1 G RMS-20-300 HZ + 3 DB/OCTAVE 300-2000 HZ; 0.02 PSD		9.3 G RMS-20-150 HZ + 6 DB/OCTAVE 150-2000 HZ; 0.045 PSD 4 MIN/AXIS DURATION	19.6 G RMS-20-60 HZ; 0.05 PSD 60-300 HZ + 3 DB/OCTAVE 300-1200 HZ; 0.25 PSD 1 MIN/AXIS DURATION
SHOCK (a)	5200 G @ 2000 HZ		3 ACTUAL SUCCESSIVE SHOCKS	3 ACTUAL SUCCESSIVE SHOCKS
ACCOUSTICS				
LAUNCH (b)	142 DB OVERALL LEVEL		146 DB OVERALL LEVEL	146 DB OVERALL LEVEL
ENTRY (a)	149 DB OVERALL LEVEL	141 DB OVERALL LEVEL	+3 DB OVERALL LEVEL	+3 DB OVERALL LEVEL

NOTE: THE DESIGN LIMITS, ULTIMATE FACTORS OF SAFETY, AND MARGINS WERE ESTABLISHED BASED ON PROBE MISSIONS THROUGH VENUS SURFACE IMPACT TO ASSURE ACCOMPLISHMENT OF SCIENTIFIC OBJECTIVES WHICH ARE SATISFIED PRIOR TO SURFACE IMPACT.

(a) OPERATING CONDITION
(b) NONOPERATING CONDITION

FOLDOUT FRAME /



EXPLANATION

THE CONCEPT OF A COMMON PROBE SHAPE FOR LARGE AND SMALL PROBES WAS ADOPTED TO REDUCE AERO TESTING AND WAS IMPLEMENTED BY SELECTING A 55° FOREBODY CONE HALF ANGLE. THIS AFFORDED A COMPROMISE BETWEEN THE HIGH C_D DESIRED FOR THE ENTRY PHASE OF THE STAGED LARGE PROBE FLIGHT PROFILE AND THE LOWER C_D DESIRED FOR THE NONSTAGED SMALL PROBE. THE 45° CONE/SPHERE AFTERBODY, WHICH HAD BEEN FOUND TO FLY WELL IN THE SPIN TUNNEL TEST ON 60° FOREBODIES (SEE SELECTED THOR/DELTA CONFIGURATION DISCUSSION), WAS PICKED AS THE COMMON AFTERBODY SHAPE OVER THE 30° CONE/SPHERE USED ON THE THOR/DELTA SMALL PROBE SINCE THE 30° SHAPE WHEN APPLIED TO THE LARGE PROBE CAUSED S/C ATTACHMENT PROBLEMS AS WELL AS INCREASING DESCENT CAPSULE SUPPORT STRUCTURE WEIGHT. THE FLARE STABILIZED DESCENT CAPSULE WAS SUPPORTED AND RELEASED AS IN THE THOR/DELTA DESIGN BY EXPLOSIVE NUTS LOCATED AT THE TOP OF THE PAYLOAD SUPPORT RING. AN ELLIPTICAL SHAPE WAS USED FOR THE SMALL PROBE PRESSURE VESSEL TO MAINTAIN A FORWARD CG (5% OF THE FOREBODY DIAMETER FORWARD OF THE MAXIMUM DIAMETER).

A FURTHER EVALUATION OF DESCENT CAPSULE INDICATED AN IMPROVEMENT IN AERODYNAMIC BEHAVIOR COULD BE OBTAINED BY USE OF A PERFORATED, THICK RING LOCATED AS SHOWN. (SEE CHAPTER 7.1) THIS DESIGN WAS INITIALLY INTEGRATED INTO THE AEROSHELL ESSENTIALLY AS THE FLARE STABILIZED DESIGN HAD BEEN. HOWEVER THE ADVENT OF A LARGER 6.6 M PARACHUTE CAUSED THE AFTERBODY TO GROW IN DEPTH AND THUS LENGTHEN THE DESCENT CAPSULE SUPPORT STRUCTURE WHICH, IN THE CASE OF THE RING STABILIZATION CONFIGURATION, WAS NO LONGER SERVING AN ADDITIONAL ROLE AS A STABILIZING FLARE.

THE SMALL AFTERBODY SHOWN WAS INTRODUCED TO TAKE FULL ADVANTAGE OF SIMPLIFICATIONS AFFORDED BY THE CENTRAL, PERFORATED RING STABILIZATION DEVICES ON THE DESCENT CAPSULE. MAJOR ADVANTAGES PROVIDED ARE:

- REDUCTION IN DESCENT CAPSULE SUPPORT STRUCTURE WEIGHT AND COMPLEXITY
- ABILITY TO USE A COMMON SEPARATION DEVICE, EXPLOSIVE NUTS, FOR BOTH FOREBODY AEROSHELL SEPARATION AND AFTERBODY AEROSHELL SEPARATION (PIN PULLERS WERE FOUND MORE ADAPTABLE FOR THE LATTER SEPARATION EVENT IN THE EARLIER CONFIGURATION.)
- ABILITY TO ATTACH PROBE TO SPACECRAFT NEAR PROBE CG AND TO ENHANCE COMMONALITY OF ORBITER AND PROBE BUS STRUCTURE
- MORE ASSURANCE OF CLEARANCE OF PROTRUDING SENSORS ON DESCENT CAPSULE UPON FOREBODY SEPARATION; HENCE LESS NEED FOR DEVELOPMENT TESTING IN THAT REGARD

IN SPITE OF THE ADVANTAGES TWO DISADVANTAGES, WHICH WERE AGGRAVATED BY THE INSTITUTION OF THE LARGER PARACHUTE, STILL REMAINED WITH THE ABOVE CONFIGURATION: AN EXCESSIVELY DEEP AFTERBODY WHICH COULD BE SUSCEPTIBLE TO FLOW REATTACHMENT AND RESULTING DEGRADATION OF STABILITY DURING ENTRY; AND A LONG-STEMMED, ASSYMMETRICAL ANTENNA INSTALLATION--BAD FOR THE ANTENNA PERFORMANCE BEFORE THE TERMINAL DESCENT PHASE AND BAD FOR THE AERO STABILITY OF THE SPHERE IN TERMINAL DESCENT. CONSEQUENTLY, A LATERALLY DEPLOYED MAIN CHUTE CONFIGURATION WAS EVALUATED THAT RELIEVED THESE DISADVANTAGES. THIS CONCEPT, HOWEVER, WAS FOUND NOT TO BE DESIRABLE DUE TO LARGE AMPLITUDE PITCHING MOTIONS INDUCED BY THE PARACHUTE UPON ITS INFLATION OUT TO THE SIDE OF PROBE (NOT DUE TO MORTAR REACTION SINCE THE MORTAR THRUST LINE IS THROUGH THE CG). THE DYNAMICS OF THIS BEHAVIOR ARE DISCUSSED IN CHAPTER 7.5. THIS CONFIGURATION IS ALSO UNDESIRABLE SINCE IT REQUIRES BALLAST WEIGHT OF AN AMOUNT ABOUT EQUAL TO THE PARACHUTE, ~10 KG.

AN ALTERNATIVE METHOD OF ACCOMPLISHING THE SAME OBJECTIVES IS TO LOCATE THE PARACHUTE IN THE FOREBODY AS SHOWN, AND TO DEPLOY IT BY MEANS OF A PILOT CHUTE LOCATED ADJACENT TO IT, OR ACROSS THE DIAMETER TO MINIMIZE THE BALLAST PROBLEM. THIS APPROACH LARGELY ELIMINATES THE PITCHING MOTIONS OF THE PREVIOUS CONFIGURATION BUT RETAINS THE BALLASTING PROBLEM AND INTRODUCES A PROBLEM WITH GETTING THE PARACHUTE LINES TO STRIP OUT PROPERLY SINCE THEY MUST CROSS THE SEPARATION JOINT BETWEEN FOREBODY AND DESCENT CAPSULE. IT SHOULD BE NOTED THAT THE BALLASTING PROBLEM IS NOT JUST ONE OF MAINTAINING THE CG BUT ALSO MAINTAINING EQUAL MOMENTS OF INERTIA IN PITCH AND YAW AND KEEPING THIS PRINCIPAL AXES OF INERTIA COINCIDENT WITH THE BODY AXES.

A BETTER LOCATION FOR A PILOT DEPLOYED PARACHUTE THAN THE ONE JUST DISCUSSED IS THAT SHOWN IN THE ACCOMPANYING SKETCH. HERE THE MASS OF THE CHUTE IS ALMOST CENTERED AND A SHALLOW AFTERBODY DEPTH IS MAINTAINED. THIS DESIGN RESULTS IN SOMEWHAT OF A DEPARTURE FROM STANDARD PARACHUTE PACKAGING TECHNIQUES BUT HAS BEEN SUCCESSFULLY EMPLOYED IN AIRCRAFT EJECTION SEAT PARACHUTE APPLICATIONS.

THE CONFIGURATION FINALLY SELECTED IS QUITE SIMILAR TO THE PREVIOUS ONE EXCEPT THAT TO ACHIEVE A RECTANGULAR PACK FOR THE MAIN CHUTE, THE ANTENNA IS LOCATED BELOW THE CHUTE PACKAGE. THIS REQUIRES AN RF TRANSPARENT CANISTER (FIBERGLASS) AND USE OF NONMETALLIC ELEMENTS IN THE PARACHUTE. THE METALLIC SWIVEL IS PACKAGED IN THE LOW PORTION OF THE CANISTER NEAR THE ANTENNA GROUND PLANE. THIS DESIGN, CONTINGENT ON ANTENNA PATTERN TESTS TO VERIFY THAT ATTENUATION BY THE PARACHUTE IS ACCEPTABLE, AFFORDS SUFFICIENT PERFORMANCE AND COST BENEFITS RELATIVE TO THE COMMON PROBE SHAPE TO OFFSET THE COST ADVANTAGES OF THAT CONCEPT IN THE AREA OF AERO TESTING. THE SMALL PROBE REVERTS TO THE OPTIMUM 45° FOREBODY/30° AFTERBODY SHAPE DETERMINED FOR IT IN SPIN TUNNEL TESTS.

THE HIGHER VELOCITIES OF THE 1978 MISSION REINFORCE THE DECISION TO USE THE 45° FOREBODY FOR THE MORE STEEPLY ENTERING SMALL PROBE. RADIATIVE HEATING RATES ARE INCREASED 50% BY THE VELOCITY INCREASE AND A 55° CONE HAS 1.97 X THE RADIATIVE HEATING RATE AS A 45° CONE.

Figure 5-15. Evolution of Preferred Atlas Centaur Design

FOLDOUT FRAME 2

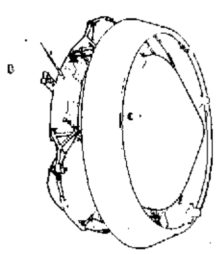
0

5.2 Probe Bus and Orbiter Configuration Tradeoffs

5.2 PROBE BUS AND ORBITER CONFIGURATION TRADEOFFS

5.2.1 Configurations Designed for Version IV Science and all Atlas/Centaur Launch for Both Missions in 1978

Minimum cost designs, which satisfy all program requirements for the 1978 Pioneer Venus missions, are defined as the preferred configurations and are shown in Figure 5-16.

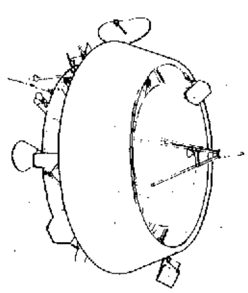


PROBE BUS

- EARTH-POINTING
- CONICAL SOLAR ARRAY
- SEQUENTIAL RELEASE
- 6.23 RAD (360 DEG) OMNIS PLUS HORN ANTENNA (AFT)
- 6-WATT TRANSMITTER

THE BASIC REQUIREMENTS ARE:

- LOW COST, STATE-OF-THE-ART, RELIABLE
- SPIN STABILIZED
- SATISFY MINIMUM DEFINED SCIENCE ACCOMMODATION VERSION IV
- 1978 LAUNCHES
- OPERATIONAL CAPABILITY USING THE 26-METER DSN SUBNETWORK
- MAXIMUM COMMONALITY BETWEEN PROBE BUS AND ORBITER
- ATLAS/CENTAUR LAUNCH VEHICLE FOR BOTH MISSIONS
- ESRO MAY PARTICIPATE IN THE ORBITER MISSION.



ORBITER

- EARTH-POINTING
- CONICAL SOLAR ARRAY
- FIXED HIGH-GAIN DISH WITH CONSCAN
- 6.23 RAD (360 DEG) OMNIS PLUS AFT HORN (PIONEERS 10 AND 11)
- 6-WATT TRANSMITTER
- GIMBALLED RAM PLATFORM

CHARACTERISTICS	PROBE BUS	ORBITER
LAUNCH DATE	LATE MAY 1978 (TYPE I)	LATE AUG 1978 (TYPE II)
LAUNCH VEHICLE PAYLOAD CAPABILITY LESS ADAPTER	782 KG (1724 LB)	508 KG (1120 LB)
SPACECRAFT WEIGHT MARGINS (KG)	111.13 KG (245 LB)	111.13 KG (245 LB)
SCIENCE PAYLOAD (KG)	13.8 KG (30.3 LB)	45.4 KG (100.1 LB)
SOLAR ARRAY POWER (WATTS)		
AT EARTH	60.4	100
AT VENUS	103	182
COMMUNICATION POWER (WATTS)	6	6
DATA RATE (BITS/S)	1024	64
THERMAL CONTROL SCHEME	PASSIVE	PASSIVE

Figure 5-16. Pioneer Venus Missions Preferred Configurations

The spacecraft configurations for both missions are based on a set of fundamental program-defined requirements plus a group of design drivers resulting from system and subsystem tradeoffs. These requirements and design drivers lead to an optional orbiter spacecraft and a set of alternative probe bus and orbiter designs based on modifying an existing spacecraft, as shown in Figure 5-17. The design drivers are discussed in Section 5.2.2; spacecraft/launch vehicle mechanical interfaces and preliminary design loads and environments are discussed in Section 5.2.6.

ALL VERSION IV
SCIENCE PAYLOAD

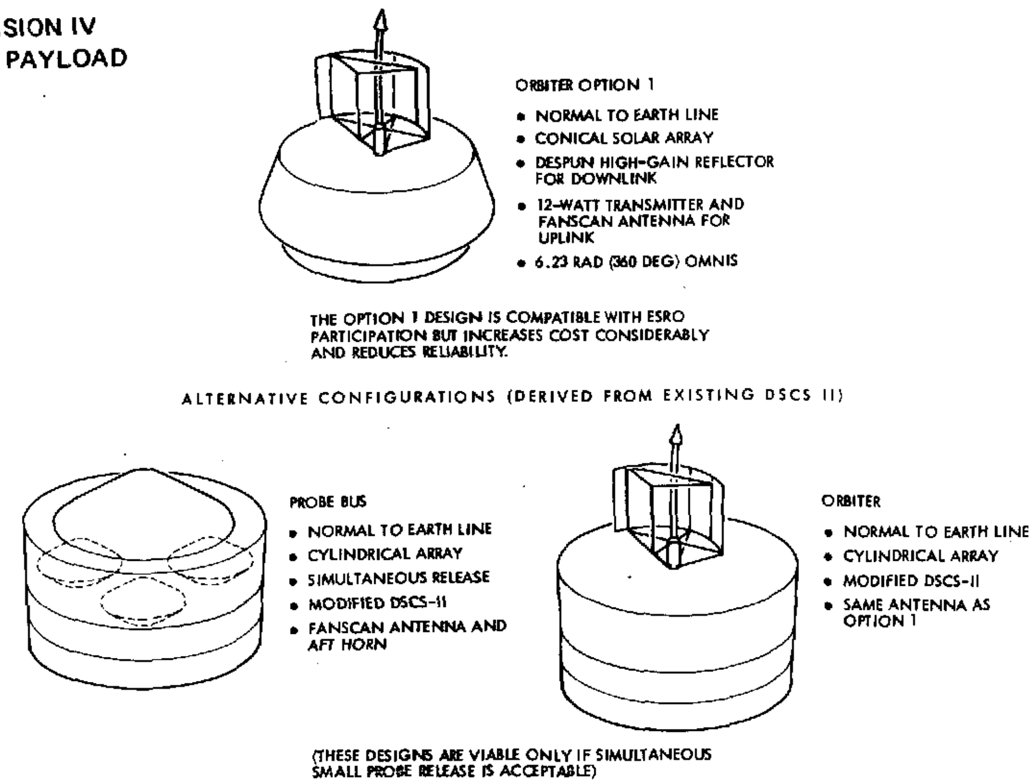


Figure 5-17. Optional Orbiter and Alternative Bus and Orbiter Configurations for Version IV Science and Atlas/Centaur Launch of Both Missions in 1978

5.2.2 Configurations Designed Under the Original Study Ground Rules

ALL VERSION III
SCIENCE PAYLOAD

The original study ground rules called for a much lower science data rate (see Sections 3.3.2 and 3.4.2), required consideration of Thor/Delta as well as Atlas/Centaur launch vehicles, and called for a 1977 probe mission launch date. On the basis of the original ground rules, we had tentatively concluded that the lowest cost program would consist of an Atlas/Centaur probe mission and a Thor/Delta orbiter mission. The configurations that were developed to meet the original requirements are shown in Figure 5-18. The development of these configurations was based on the same design drivers as those referred to in Section 5.2.1. These design drivers are considered in some detail in Section 5.2.3.

Table 5-4 presents the estimated cost variations that would result from selection of various configurations other than the preferred earth-pointing designs.

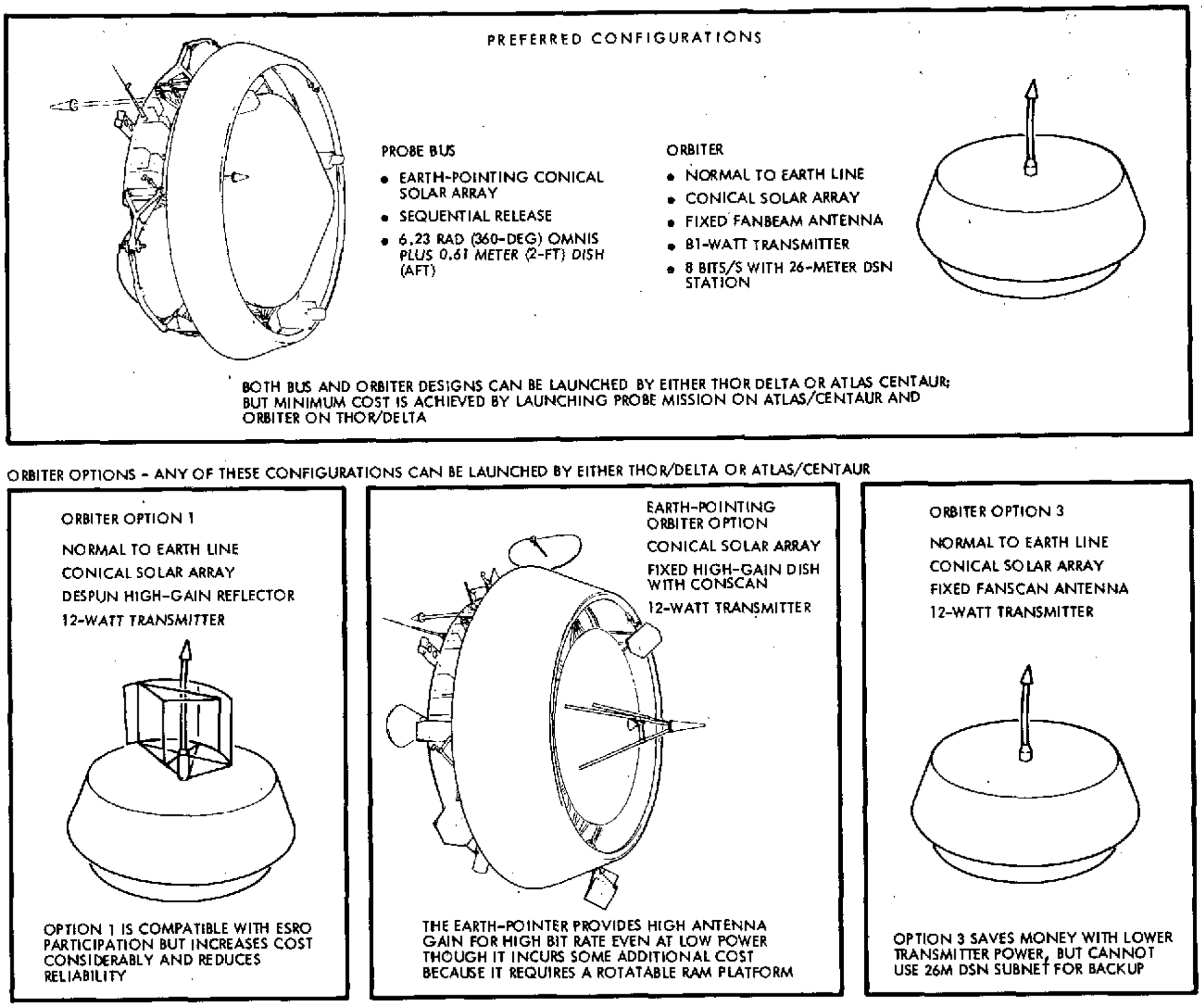


Figure 5-18. Study Configurations developed prior to Redirection for Version IV Science and 1978 Launches on Atlas/Centaur. These Low-Cost, Low-Data-Rate Configurations are Viable with Pre-Version IV Science

Table 5-4. Added Cost of Configurations Other Than the Preferred Earth-Pointers

PROBE BUS	ORBITER	COST
PREFERRED EARTH-POINTER	PREFERRED EARTH-POINTER	BASELINE
PREFERRED EARTH-POINTER	SPIN AXIS PERPENDICULAR AND DESPUN REFLECTOR	BASELINE PLUS $1,300,000
DSCS-II ADAPTATION WITH SPIN AXIS PERPENDICULAR	DSCS-II ADAPTATION WITH FANSCAN ANTENNA AND SPIN AXIS PERPENDICULAR	BASELINE PLUS $210,000

NOTE: THE DSCS-II ADAPTATION IS LIMITED TO SIMULTANEOUS RELEASE OF THE SMALL PROBES. FURTHER COSTS WOULD BE INCURRED IF IT WERE ADAPTED FOR SEQUENTIAL RELEASE.

ALL VERSION III SCIENCE PAYLOAD

The two other orbiter configurations shown in Figure 5-16 are the perpendicular spinner with a 31-watt transmitter and fanscan antenna (which would have been preferred for the pre-Version IV science payload with its low data rate requirement) and the 12-watt version of this configuration (shown in Figure 5-18 as Option 3). Of these, the 31-watt design was estimated to cost about the same as the preferred earth-pointer and the 12-watt version was estimated to cost about $140,000 less. Neither of these configurations, however, could provide the high data rate required for the Version IV payload.

5.2.3 Design Drivers

ALL CONFIGURATIONS

Each of the following configuration design drivers was examined to arrive at the most cost effective combination in terms of the spacecraft designs. Together with other tradeoffs (described in the subsystem definitions of Section 8) that have less direct effect on the configurations, these design drivers form the basis for the spacecraft system configurations. This section presents an overview of the detailed work that was done during these tradeoff studies.

5.2.3.1 Spin-Axis Orientation

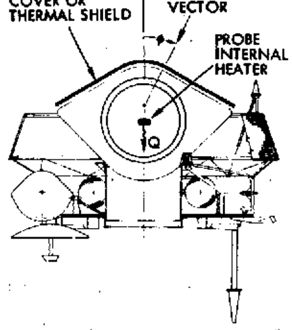
The relative positions of the sun, earth, and Venus are the fundamental considerations that affect the choice of nominal spin axis orientation for which there are three hasic alignments: earth-pointing, normal to the earth and sun lines, and normal to the spacecraft orbit about Venus.

The choice of spin axis orientation affects overall science accommodation, choice of antenna, and design of the solar array. For the probe mission, large probe thermal control is also a major consideration, as shown below.

Probe Bus

Before Venus encounter the demands on the communication subsystem are minor in terms of both power and antenna gain. Most of the science is inactive during transit so its power requirements are small. Attitude control requires little power and thermal control needs none, except for heater requirements associated with the basic alignment choice.

The small probes can be thermally controlled by insulation that can be moved out of their paths before probe release. The large probe, however, constitutes a major radiating surface which would require either heaters or a jettisonable thermal cover, as shown in Figure 5-19. This problem can be solved by orienting the spin axis toward earth so that the large probe can be pointed toward the sun most of the time for solar heating. There is an early period during transit, however, when electrical heating would be required unless the spacecraft is maneuvered to keep the solar heat input at a proper level. The conical array makes this possible, as shown in Section 5.2.2.3. During this period, the spacecraft would communicate through the omni antenna.



(ZERO COMPONENT POWER DISSIPATION WITHIN PROBE)

METHOD	SOLAR ASPECT ANGLE [RAD (DEG)]	HEATER REQUIRED	DEPLOYABLE SHIELD	REMARKS
1. UNHEATED PROBE	1.57 (90)			TOO COLD (-60°C/-85°C)
2. HEATED PROBE	1.57 (90)	X		EXCESSIVE POWER (27 WATTS NEAR EARTH)
3. INSULATION COVER	1.57 (90)	X	X	ADDED WEIGHT AND COMPLEXITY
4. THERMAL SHIELD	1.57 (90)		X	ADDED WEIGHT AND COMPLEXITY
5. EXPOSED PROBE	1.57 (90)	X		LARGE OPERATING TEMPERATURE RANGE, UNACCEPTABLE SPACECRAFT INERTIA RATIO
6. EARTH POINTER	VARIABLE			RECOMMENDED DESIGN

METHOD 1 NOT ACCEPTABLE (DOES NOT MAINTAIN LARGE PROBE TEMPERATURE). METHODS 2, 3, 5 REQUIRE HEATERS AND THEREFORE INCREASED SOLAR ARRAY AND INCREASED BATTERY POWER FOR MANEUVER WHEN THE SUN ANGLE DEPARTS SIGNIFICANTLY FROM 1.57 RADIANS (90 DEGREES). METHOD 4 ADDS WEIGHT (NOT CRITICAL FOR ATLAS/CENTAUR, BUT HIGHLY CRITICAL FOR THOR DELTA) AND MECHANICAL COMPLEXITY FOR MOUNTING, LOCKING AND EJECTING THERMAL COVER) THAT RULES IT OUT FOR BOTH LAUNCH VEHICLES. METHOD 6 IS RECOMMENDED; IT MAINTAINS LARGE PROBE TEMPERATURE BY SEVERAL MANEUVERS AFTER FIRST 55 DAYS OF MISSION AND BY EARTH POINTING THEREAFTER TO MAINTAIN APPROPRIATE TEMPERATURE CONTROL.

Figure 5-19. Effect of Spin-Axis Orientation on Thermal Control of Large Probe

The recommended scheme is a modified earth pointer, which permits control of spin axis pointing to eliminate the need for any heater power and yet maintains required antenna orientation throughout the mission. The solar aspect angle would vary from a minimum of 0.21 radians (12 degrees), to a maximum of 0.72 radians (41 degrees) at large probe release. Further details are shown in Section 8.7.

Orbiter

The spin axis orientation preferred for the orbiter is earth pointing because it permits a fixed high-gain antenna similar to the Pioneer 10 and 11 design. The ion and neutral mass spectrometers would have to be mounted on a deployable gimballed platform to achieve the proper ram angle. This adds some complexity to science data reduction and to ground control operations for periodic updating during orbital operation to maintain ram pointing angles. The large fixed reflector, however, permits substantial growth in science data rate.

For the Version III science payload, the science data rate was low enough that a fanbeam/fanscan antenna configuration was adequate; then the recommended spin axis orientation would have been normal to the earth line and sun lines (except near syzygy, when an attitude normal to earth line and nearly normal to the orbit plane of Venus is maintained). This simplified data reduction and eliminated the need for a ram platform. With the much higher data rates, and with a normal spin axis, however, a large despun antenna (with attendant cost and reliability degradation) would be required.

On the basis of cost, then, the earth pointing orientation is recommended for the orbiter.

As indicated in Figure 5-17, however, an existing TRW spacecraft could be adapted for these missions. This would have to be oriented normal to the earth line, and would also require simultaneous small probe release. This is described in Section 5.2.3.4. It was studied on the basis that adaptation of an existing design might result in the lowest cost configuration. This turns out not to be the case, and there are other undesirable features, as will be seen.

A configuration with its spin axis normal to the plane of the spacecraft orbit about Venus would require a double-gimballed antenna, which would further increase cost and complexity and decrease reliability. This approach was, therefore, rejected. It was used with great success for the Atmospheric Explorer but, in this application, high data rate communication could be achieved with a low-gain antenna in earth orbit.

5.2.3.2 Antenna Selection ALL PROBE CONFIGURATIONS

Probe Bus

As stated in the previous section, the choice of antennas is less a design driver for the probe bus configuration than a logical consequence of the choice of spin axis orientation. Given the selection of an earth pointer (with variable orientation for large probe thermal control) the choice of antennas is clear: two flight-proven omni antennas, one pointing forward and one aft, provide 6.28-radian (360-degree) communication coverage through the second midcourse maneuver, and a proven medium-gain horn fulfills all communication requirements for the remainder of the mission.

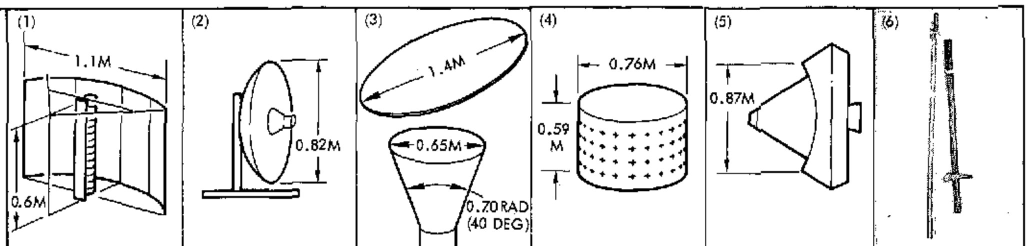
If the spin axis is normal to the earth line (as it would be for the alternative configuration) an additional medium gain, fanbeam antenna would be required for downlink and a fanscan antenna for uplink. This adds appreciably to the cost.

Orbiter ALL ORBITER CONFIGURATIONS

For a design with spin axis normal to the earth line, a fanbeam Franklin array (see Figure 5-20) is the lowest cost antenna configuration. It is the same device as that used on Pioneers 6 through 9. Since it is fixed, it interfaces simply and directly with the spacecraft subsystems but its modest gain capability does call for considerable transmitter power to work with the 26 meter DSN subnet even with the relatively low data rates of the initial study guidelines. Adding a fanscan section provides a precise attitude determination capability.

With the higher data rates required for science by the 13 April redirection, however, antenna selection does become a design driver; the fanbeam is no longer a viable alternative. The large, earth-pointing, fixed reflector (similar to the dish used on Pioneers 10 and 11) provides a high-gain, low-cost, reliable solution of the problem with low transmitter power requirements. This is shown in our preferred configuration.

A despun reflector fed by a Franklin array, similar to the system used on Helios, would also meet higher data transmission requirements



ANTENNAS / CHARACTERISTICS	(1) DESPUN PARABOLIC CYLINDER	(2) DESPUN PARABOLIC ANTENNA	(3) DESPUN FLAT PLATE REFLECTOR	(4) ELECTRONICALLY DESPUN ARRAY	(5) FIXED PARABOLOID (⊕ POINTING)	(6) FRANKLIN ARRAY
ANTENNA GAIN (DB)	22.5	22.5	21	22	23.5	11
WEIGHT PENALTY, KG (LB)	9.5 (20.9)	10.9 (24.0)	15.6 (34.3)	6.4 (14.1)	0	2.0 (4.6)
PRIMARY POWER (WATTS)	56	56	78	46	37	37
RELIABILITY	0.986	0.985	0.986	0.965	0.999	0.999
DEVELOPMENT STATUS (MAJOR ITEMS)	DEA MODS REQUIRED	DEA MODS REQUIRED	• ANTENNA DEVELOPMENT • DMA AND DEA MODS REQUIRED • REQUIRES CAGING	STUDY CONTRACT	SCALED DOWN PIONEERS 10 AND 11 REFLECTOR	FLOWN ON PIONEERS 6 THROUGH 9
COST PENALTY COMPARED TO NO. 6 ($M)	0.75	0.77	0.97	0.80	0	0.15

Figure 5-20. Orbiter Antenna Tradeoffs

but would impose a cost and reliability degradation implicit in the addition of a despin mechanism. A despun horn/flat-plate reflector combination (as used on Intelsat III) is physically large and unwieldy and the electronically despun array constitutes a new development, which is risky from both a cost and schedule standpoint. A despun parabolic dish, with integral despun feed, would also meet requirements but would bring with it the additional interface and reliability problems of a rotary RF joint and slip rings. This would offer no advantages, as compared with the Helios approach, so the despun reflector fed by the Franklin array would be our optional recommendation. This of course would accompany a normal to earth line spin axis orientation. In all cases, it is easier to incorporate an X-band link (for the dual frequency occultation experiment) on the earth pointer configuration.

5.2.3.3 Solar Array

ALL CONFIGURATIONS

Both the configuration of the solar array (conical or cylindrical) and its location (above or below the center of mass or split and mounted above and below) affect spacecraft design and performance.

The configuration tradeoff, as indicated earlier in this section, favors the conical design for several reasons. Figure 5-21 provides

ALL CONFIGURATIONS

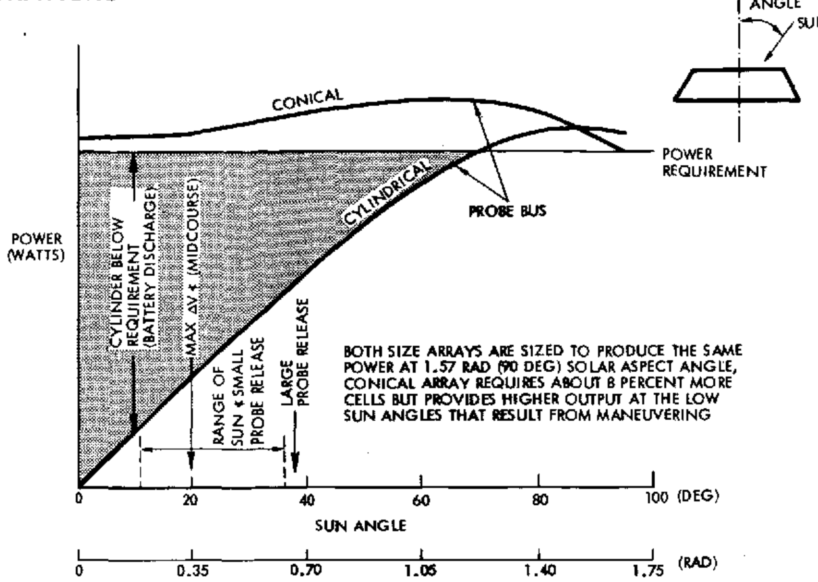


Figure 5-21. Comparison Between Power Outputs of Conical and Cylindrical Arrays

comparative performance data relating to this choice. For the comparison, both configurations were sized to produce the same power at a solar aspect angle of 1.66 radians (95 degrees). Although the conical array requires about 8 percent more cells than the cylindrical form, its output is higher at the low sun angles that result during maneuvers. This gives the flight operations crew freedom to maneuver without haste through a wide range of attitudes without concern for battery power management and saves on overall array size by eliminating the need for power to heat the large probe.

Both probe bus and orbiter have to maneuver at low sun angles. Specifically, the solar aspect angles for the bus are between 0.21 and 0.70 radians (12 and 40 degrees) during probe release. The probe bus and orbiter maneuvers for midcourse corrections can be made at almost any angle. Consequently, we recommend a conical array for both missions.

<u>Probe Bus</u>

ALL PROBE CONFIGURATIONS

The study shows that a 0.39 radian (22.5 degree) cone angle provides a good compromise between the needs of the probe and orbiter missions as well as providing hardware commonality (see Section 8.1).

A cylindrical array could be used, if the spin axis were normal to the earth and sunlines, but this (as indicated in Section 5.2.3.1) would require either a large, heavy, jettisonable thermal cover for the large probe or it would require heaters. To provide additional power for the heaters, for example, would require a 50 percent increase in array size alone. This approach is, therefore, not recommended.

To keep the array cool (and thereby minimize the number of solar cells for a given power output), it is mounted above the equipment compartment so that it can radiate to space. Decoupling it in this way from the equipment compartment thermal control system also permits greater freedom in mounting equipment and science instruments. The forward location also facilitates adjustment of the center of mass below the solar array and so facilitates mounting the small probes in the plane of the center of gravity for sequential release. It does introduce some attitude drift due to light-pressure but this is small [about 0.001 radian (0.1 degree)per day or less for the preferred configuration].

Orbiter

With the spin axis normal to the Venus orbit plane, a cylindrical array would be adequate for the orbiter nut would not provide the operational flexibility during maneuvers that is provided by the conical configuration. Nor would it maintain hardware commonality between the two missions. This configuration could be converted to an earth pointer, however, simply by replacing the fixed parabolic reflector with a despun reflector; the additional power could easily be provided by increasing the depth of the solar array panels.

The forward location of the array does make it more susceptible to attitude drift due to solar pressure, particularly for the Thor/Delta versions, because the center of pressure of the spacecraft configuration is displaced from its center of mass. This drift could be removed by splitting the array, with a forward portion (identical to the probe bus array) and an added after-portion around the equipment compartment. This would eliminate attitude drift by balancing out the light pressure but the aft array would have to be back-insulated. With conventional array packing factors, this would raise solar cell temperatures to near their

upper limits. So a reduced packing factor and conducting substrates would be necessary to provide an acceptable cell temperature margin. A further complication of the split array is that it makes it difficult to provide acceptable viewing angles for the spectrometer instruments.

5.2.3.4 Sequential Versus Simultaneous Small Probe Release

Both methods of probe release are capable of separating the small probes for geographical dispersion upon arrival at Venus. With simultaneous release, geographic separation is achieved by increasing the spin rate. The higher it is, the greater the separation. Sequential release, however, permits better control of separation both spatially (for geographical dispersion) and in time (for the added advantage of redundant communications coverage by the DSN).

This capability for careful targeting (and, if necessary, retargeting) also provides the capability for entry at or near zero angle of attack, which reduces wobble. This, in turn, facilitates determination of atmospheric density on the basis of data from the axial accelerometer, which is the only means of estimating the density, and hence the composition, of the upper atmosphere.

Sequential release does require that the centers of mass of the small probes be in the same plane as the composite center of mass of the bus after large probe release. The release of each small probe then causes spin axis translation without nutation, or tipping of the spin axis with respect to the mechanical axis.

This constraint on spacecraft center of mass requires the center of mass of all expendables to be located at the station common with the probe centers of mass. This requires an equipment platform for mounting the small probes, expendables, and most spacecraft equipment, and is consistent with sequential release. The need to maintain the inertia ratios of the bus ≤ 1.10 at all times also drives the mounting stations for the small probes and most spacecraft equipment close together. This is discussed in Section 6.2.

5.2.4 Probe Bus Configuration Details

Probe bus spacecraft were configured for both launch vehicles. The configurations all use a central cylinder to interface with the launch

vehicle and to support the large probe. An annular equipment platform is supported by the central cylinder and by a truss system. The small probes are supported from the platform, with their centers of mass in the same plane as the spacecraft composite center of mass after large probe release in order to allow sequential release. Most of the bus science instruments and spacecraft subsystem components are mounted to the platform. The conical solar array is supported by the truss system.

The spin axis orientation selected for each probe bus is nominally earth pointing, to maintain large probe thermal levels during transit, as described in Section 5.2.3.1.

5.2.4.1 Thor/Delta Probe Bus Configuration T/D III

The probe bus configuration for the Thor/Delta launch vehicle is shown in Figures 5-22 and 5-23, and a weight summary in Table 5-5. This configuration meets all initial requirements but, in order to provide an adequate weight contingency, several weight-saving items are incorporated in the configuration. These include a silver-zinc battery and a beryllium central cylinder, which provide sufficient contingency but add to the cost.

5.2.4.2 Atlas/Centaur Preferred Probe Bus Configuration A/C IV

The Atlas/Centaur probe bus configuration, Figures 5-24 and 5-25, is similar to the Thor/Delta bus but scaled up to accommodate the larger Atlas/Centaur probes and to be compatible with the Centaur launch vehicle stage and its fairing. The weight statement (Table 5-5) shows an adequate contingency between the estimated requirement and the net launch vehicle capability for this mission. The larger Atlas/Centaur probes present very significant cost advantages over the smaller Thor/Delta probes. In addition, the larger platform area permits easier subsystem and science component installation on the probe bus. For these reasons, plus the ample contingency and its capability for additional science, the Atlas/Centaur bus is the preferred over the Thor/Delta for the probe mission. This preference has now, of course, become a requirement as a result of the 13 April redirection.

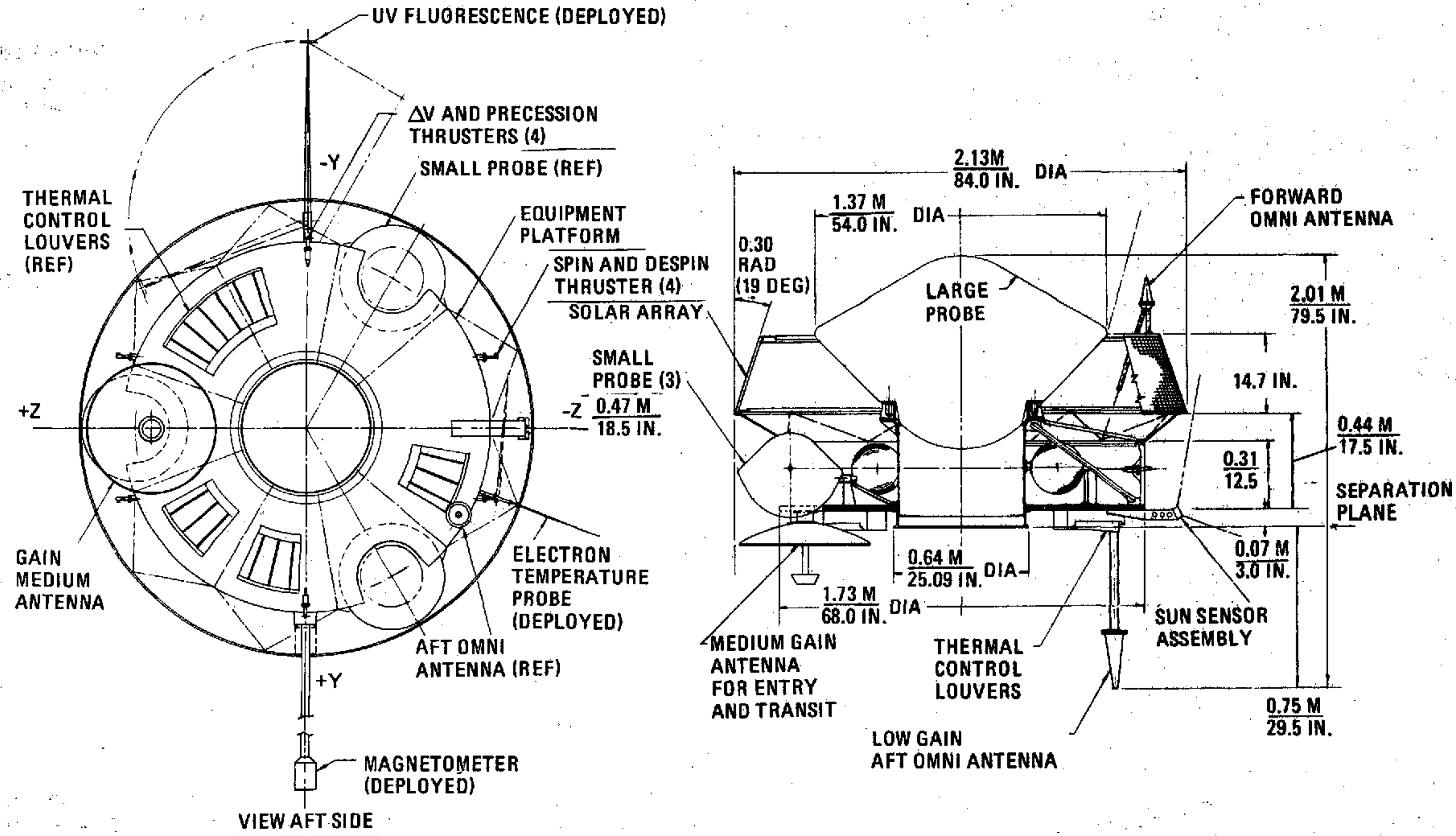


Figure 5-22. Recommended Thor/Delta Probe Bus

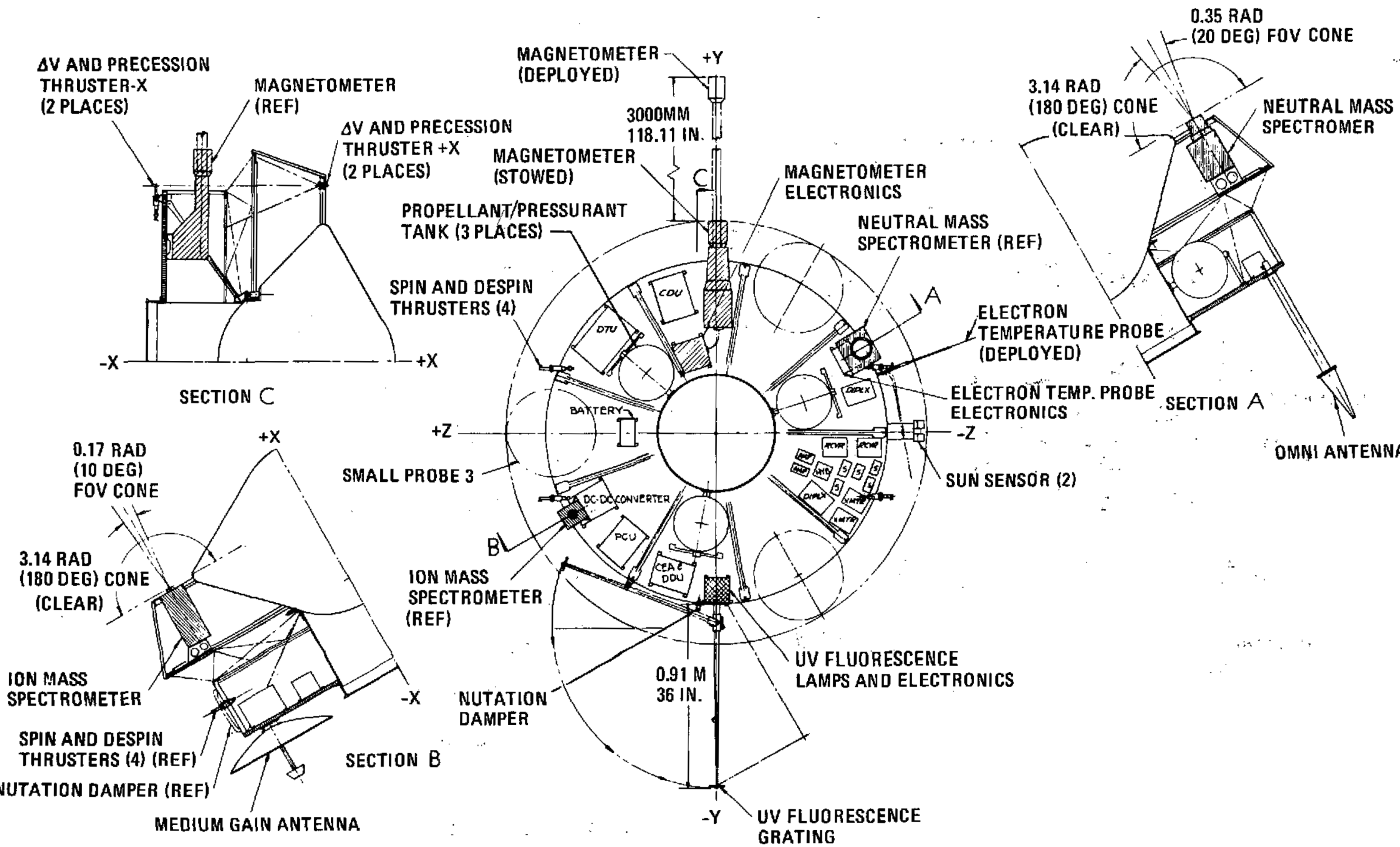
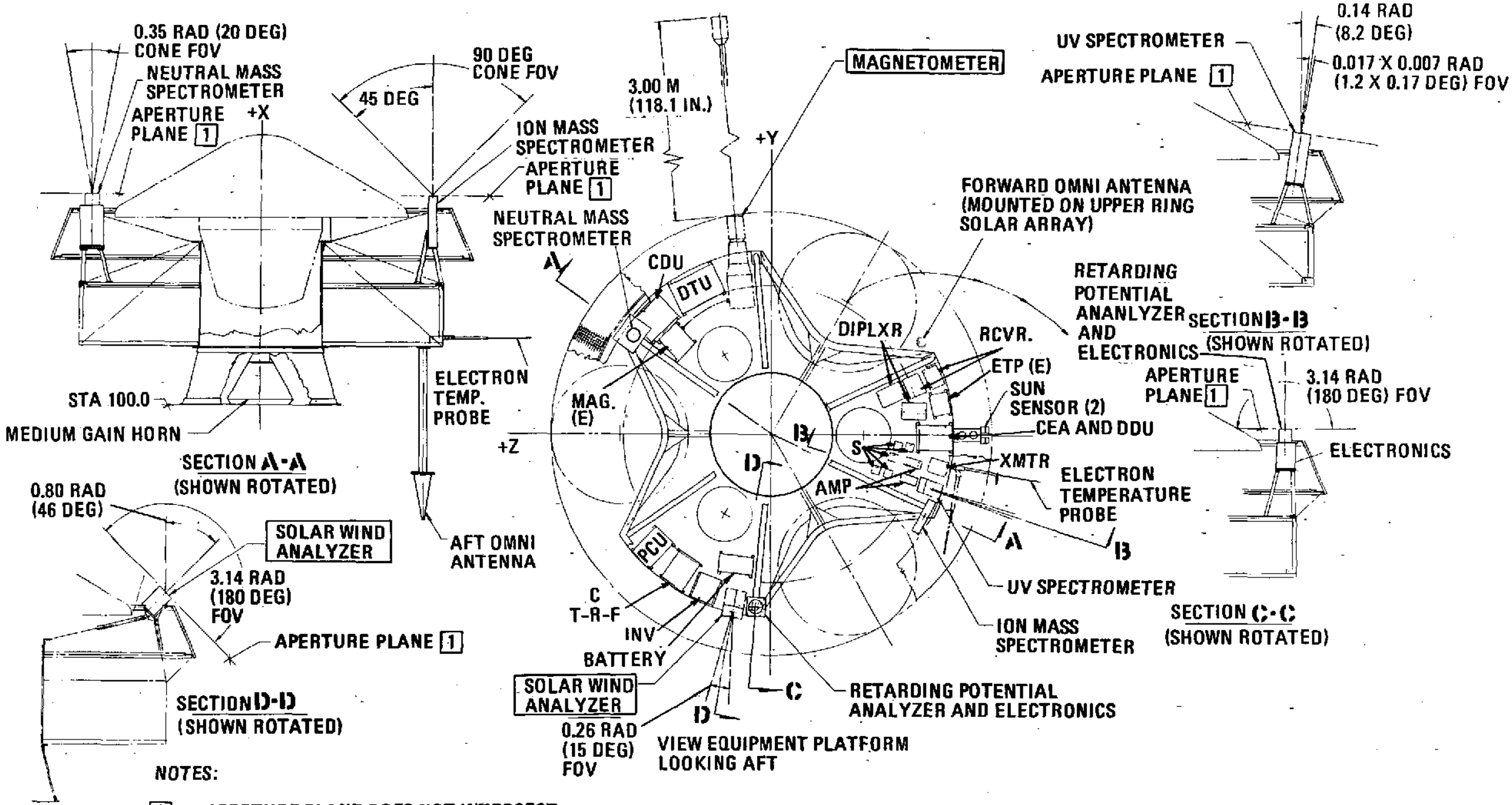


Figure 5-23. Thor/Delta Probe Mission Science and Equipment Layout



NOTES:

1. APERTURE PLANE DOES NOT INTERSECT ANY PART OF SPACECRAFT BUS THEREFORE EMISSIONS FROM SPACECRAFT OR THRUSTERS CANNOT ENTER DIRECTLY INTO APERTURE.

2. ITEM DENOTES OTHER CANDIDATE INSTRUMENTS.

Figure 5-24. Atlas/Centaur Probe Mission Science and Equipment Layout With Other Candidate Instruments

Page intentionally left blank

A/C IV T/D III

Table 5-5. Probe Spacecraft Configuration Weight Summary

DESCRIPTION	WEIGHT			
	THOR/DELTA [KG (LB)]		ATLAS/CENTAUR [KG (LB)]	
ELECTRICAL POWER	15.7	(34.7)	21.5	(47.4)
COMMUNICATIONS	9.4	(20.8)	13.2	(29.1)
ELECTRICAL DISTRIBUTION	12.2	(26.8)	15.5	(34.1)
DATA HANDLING	3.9	(8.5)	3.9	(8.5)
ATTITUDE CONTROL	2.3	(5.1)	2.7	(6.0)
PROPULSION (DRY)	6.8	(14.9)	7.9	(17.4)
THERMAL CONTROL	10.4	(23.0)	15.5	(34.3)
STRUCTURE	44.1	(97.2)	75.4	(166.2)
BALANCE WEIGHT PROVISION	2.7	(6.0)	5.4	(12.0)
PROBE BUS LESS SCIENCE (DRY)	107.5	(237.0)	161.0	(355.0)
SCIENTIFIC INSTRUMENTS – BUS	11.2	(24.6)	13.8	(30.3)
PROBE BUS (DRY)	118.7	(261.6)	174.8	(385.3)
PROBES	222.2	(489.8)	473.7	(1044.4)
SPACECRAFT (DRY)	340.9	(751.4)	648.5	(1429.7)
HYDRAZINE PROPELLANT AND PRESSURANT	19.4	(42.8)	19.9	(43.8)
SPACECRAFT LESS CONTINGENCY	360.3	(794.2)	668.4	(1473.5)
CONTINGENCY (NET ALLOWABLE)*	24.8	(54.8)	113.6	(250.5)
	(7.3%)		(17.5%)	
GROSS SPACECRAFT AFTER SEPARATION	385.1	(849.0)	782.0	(1724.0)

*PERCENTAGE VALUES ARE RELATIVE TO THE DRY SPACECRAFT WEIGHT.

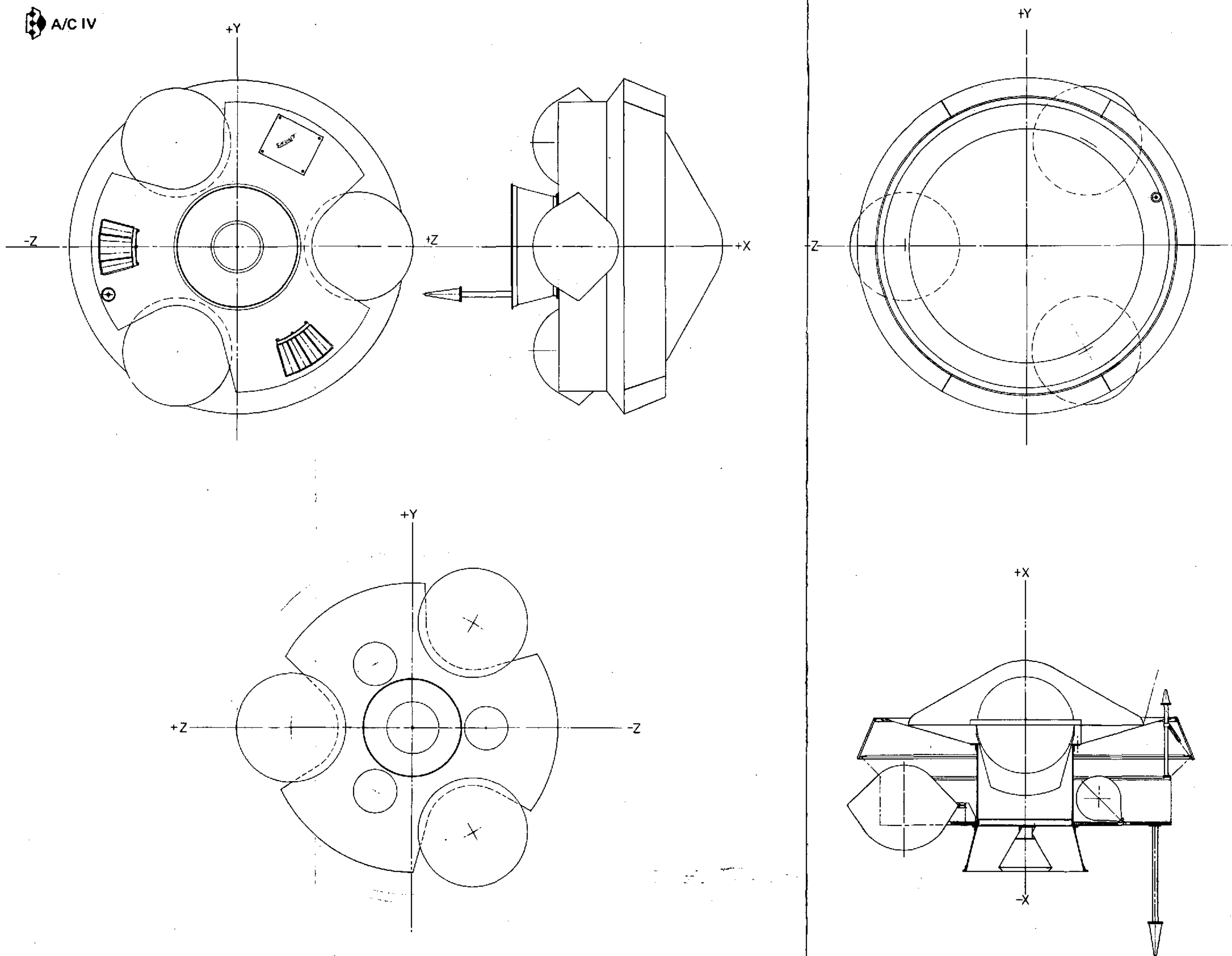


Figure 5-25. Preferred Atlas/Centaur Probe Mission Spacecraft

5.2.4.3 Alternative Design Based on DSCS-II Spacecraft

Several existing spacecraft configurations were examined for adaptation to a Pioneer Venus design, but none proved as cost effective as the preferred design. The DSCS-II spacecraft at first appeared as an excellent candidate for conversion but detailed examination showed that the modifications required to convert this communications satellite (which is designed for operation at synchronous orbit) for the Pioneer Venus mission compromised performance and cost more than designing a simple, reliable, flexible pair of spacecraft specifically for the Venus missions.

Figure 5-26 shows the probe mission spacecraft based on DSCS-II. Notably, only a fraction of its solar array area is required, and accommodations for the probes require extensive structural changes and additions. A new launch vehicle/spacecraft adapter is required to support the four spacecraft attachment points to the continuous Intelsat IV adapter at the forward end of the Centaur.

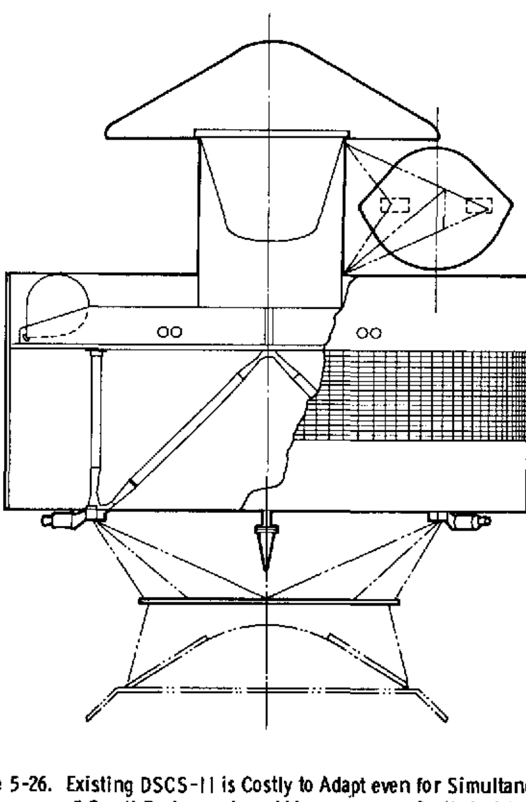


Figure 5-26. Existing DSCS-II is Costly to Adapt even for Simultaneous Release of Small Probes and would be even more Costly to Adapt for Sequential Release

Only if we are willing to sacrifice sequential small probe release, can the DSCS-II be converted for the probe mission. However, the structure is considerably heavier (note that the DSCS-II spacecraft was designed to support another identical vehicle forward of it), and the resulting weight contingency for the Atlas/Centaur for the probe mission is very small (~1.1 percent). Even for an existing design, this is not adequate. For the orbiter, the corresponding contingency would be 7 percent.

The sacrifice of sequential release, the relative difficulty of attaining desired instrument fields of view and mounting some spacecraft equipment, the lack of growth, and above all the cost of this adaptation, lead us to select the earth-pointing configurations. Throughout the study, as we considered design changes to eliminate each of the adaptation's disadvantages, we were quickly led to the preferred bus design of Figure 5-24.

5.2.5 Orbiter Configuration Details ALL ORBITER CONFIGURATIONS

The configuration tradeoff discussions of Section 5.2.1 indicated that more than one combination of configuration drivers merits careful consideration. Therefore several orbiter spacecraft configurations have been formulated: a preferred design, and three options using different combinations of spin axis orientation and antennas. The fundamental drivers for each launch vehicle are identical but, by taking advantage of the Atlas/Centaur's higher payload weight capability, we have been able to use less expensive components. Only the preferred configuration and one option, however, are viable with the Version IV science payload.

Each orbiter configuration has been designed to maintain maximum commonality with the corresponding probe bus for each launch vehicle candidate. The common elements are:

- Launch vehicle interface
- Central cylinder, accommodating the orbit insertion motor and supporting the antenna assembly
- Equipment platform, with a different set of cutouts and inserts
- Solar array and platform supporting truss

- Spacecraft subsystem components
- Solar array conical angle.

The primary orbiter configuration tradeoffs for a given launch vehicle involve its spin axis orientation and communications subsystem. The power subsystem (and, therefore, the solar array requirement) is derived from these selections.

5.2.5.1 Thor/Delta 

Normal to Earth Line

With the spacecraft spin axis normal to the earth and sun lines, the initially specified orbiter science system requirements are satisfied without the need for a gimballed ram platform. The spin axis orientation impact on science is described in Section 3.2.

The difference between the several normal-to-earth-line configurations is in their arrays of antennas and power amplifier. Figure 5-27 shows a design utilizing a despun reflector antenna, similar to the Helios design. Forward and aft omni antennas, and a cocked fanbeam antenna (for fanscanning) complete the antenna complement. A conical solar array retains commonality between the orbiter and probe bus arrays and also allows off-normal operation without concern over power management. This configuration, designated as Option 1, provides a high data rate, conscan capability for attitude determination, and omni antennas for all attitude communication during maneuvers. It also uses a 12-watt solid state transmitter.

With the 64-meter subnetwork, it is feasible to reduce the spacecraft communications capability by replacing the despun reflector antenna with a fanbeam antenna, similar to the device used on Pioneers 5 through 9. A tilted fanbeam, for fanscan, and forward and aft omni antennas complete the antenna complement (Figure 5-28). The 12-watt transmitter is retained. This configuration, designated as Option 3, has the smallest power requirement and therefore the smallest array, which reduces susceptibility to solar pressure torque (due to the equivalent center of pressure being displaced from the center of mass; see Section 8.5). This

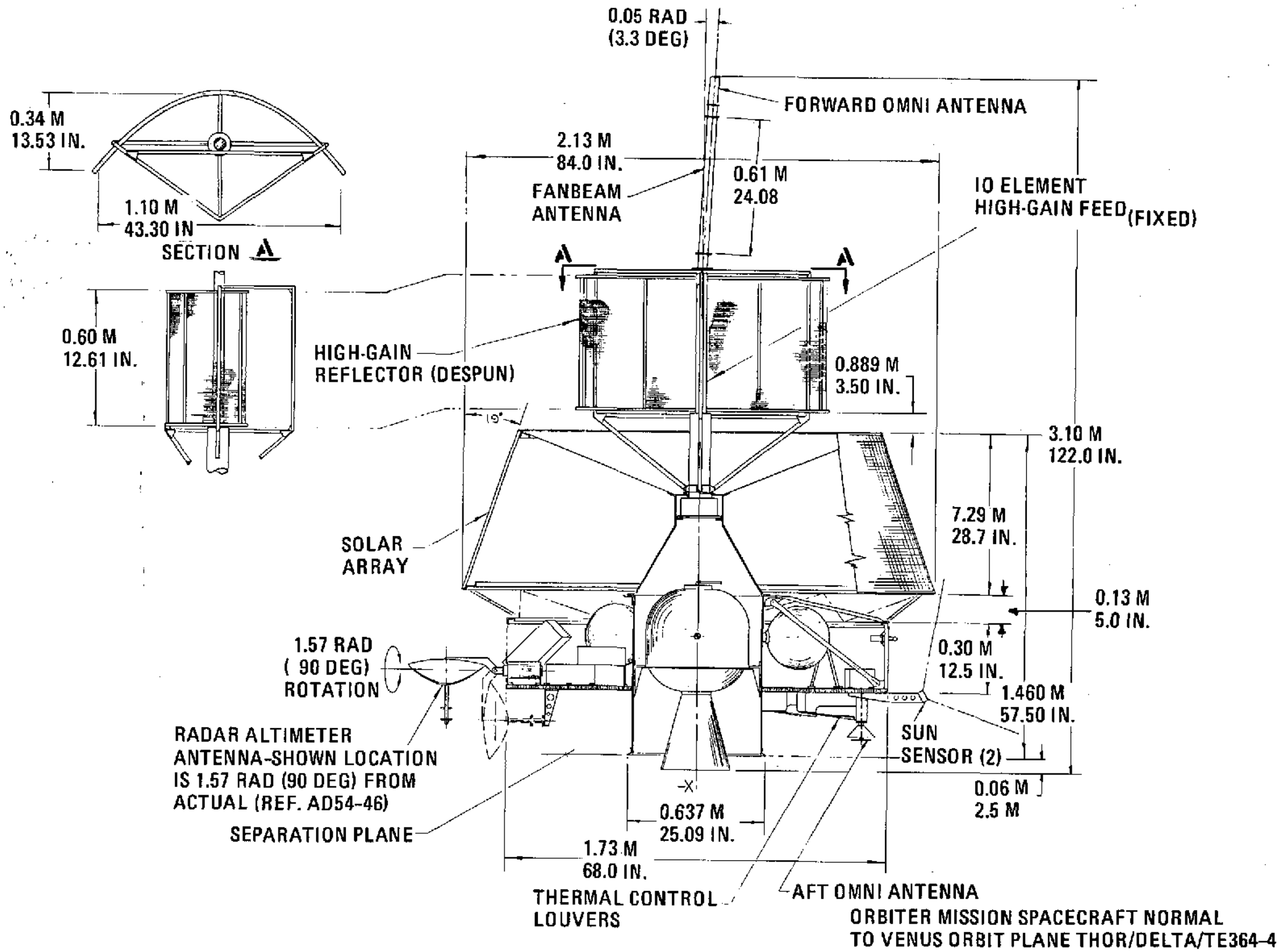


Figure 5-27. Thor/Delta Orbiter Despun Reflector and Fanscan

T/D III

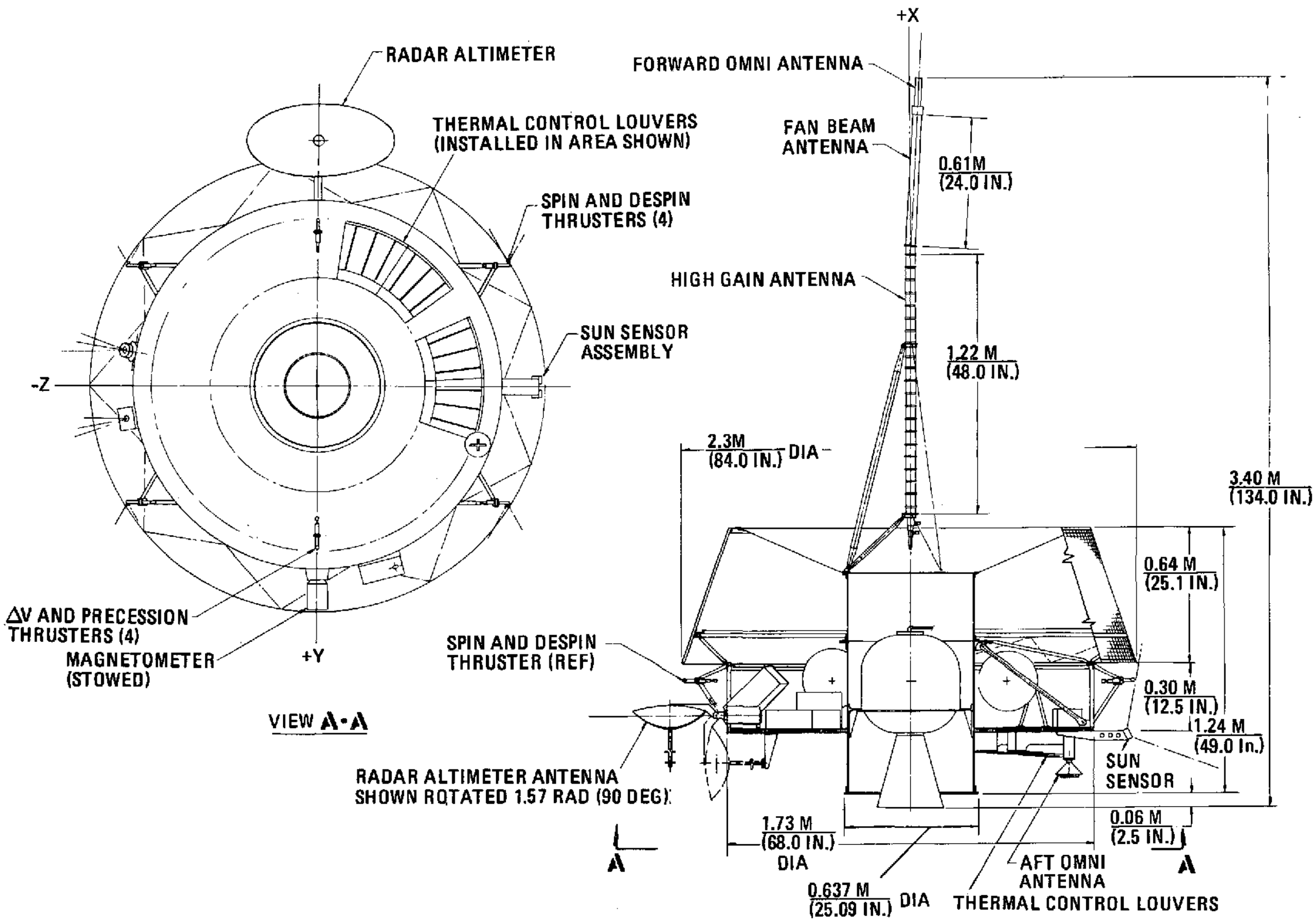


Figure 5-28. Thor/Delta Orbiter 12-Watt Fanbeam/Fanscan

T/D III

configuration is the least expensive of the Thor/Delta orbiter designs and shows the highest allowable weight contingency.

12 W T/D III

31 W T/D III

By increasing the power of the transmitter, the fanbeam/fanscan configuration can be made capable of using the 26-meter DSN subnetwork. A 31-watt TWTA transmitter with the fanbeam antenna meets this requirement; this Option 2 spacecraft is shown in Figure 5-29. Its solar array is enlarged for the higher power transmitter. The higher power TWTA is more efficient than the solid state transmitter, but a heat sink is required to maintain acceptable TWT temperatures. The TWTA itself is also significantly heavier than the solid state device.

In order to maintain a weight contingency similar to the 12-watt fanscan configuration of Figure 5-28, it was necessary to incorporate several changes that further increased the cost of this design. These include incorporation of a silver-cadmium battery instead of nickel-cadmium, and a beryllium central cylinder. Because this is the minimum spacecraft that meets all program requirements, it was the preferred Thor/Delta configuration before the Version IV science requirements were announced.

Another possible configuration incorporated an electrically despun antenna instead of the mechanically despun reflector of Option 1. This is shown in Figure 5-30. Because of its higher development cost and risk, this arrangement was not carried any further.

T/D III

Earth Pointer. For an earth pointing spin axis, a fixed high-gain parabolic antenna can be used to provide high gain and high bit rate telemetry during Venus orbit (Figure 5-31). The communication subsystem is based on a 12-watt solid state transmitter. An offset feed provides conscanning, and the aft reflector antenna provides communications and conscan functions during transit. A conical solar array supplies spacecraft power.

This design is probably the only Thor/Delta version which could be upgraded to carry the Version IV science compliment while retaining an adequate weight margin. A deployable, gimballed experiment platform is required to achieve the proper ram angle required by the ion and

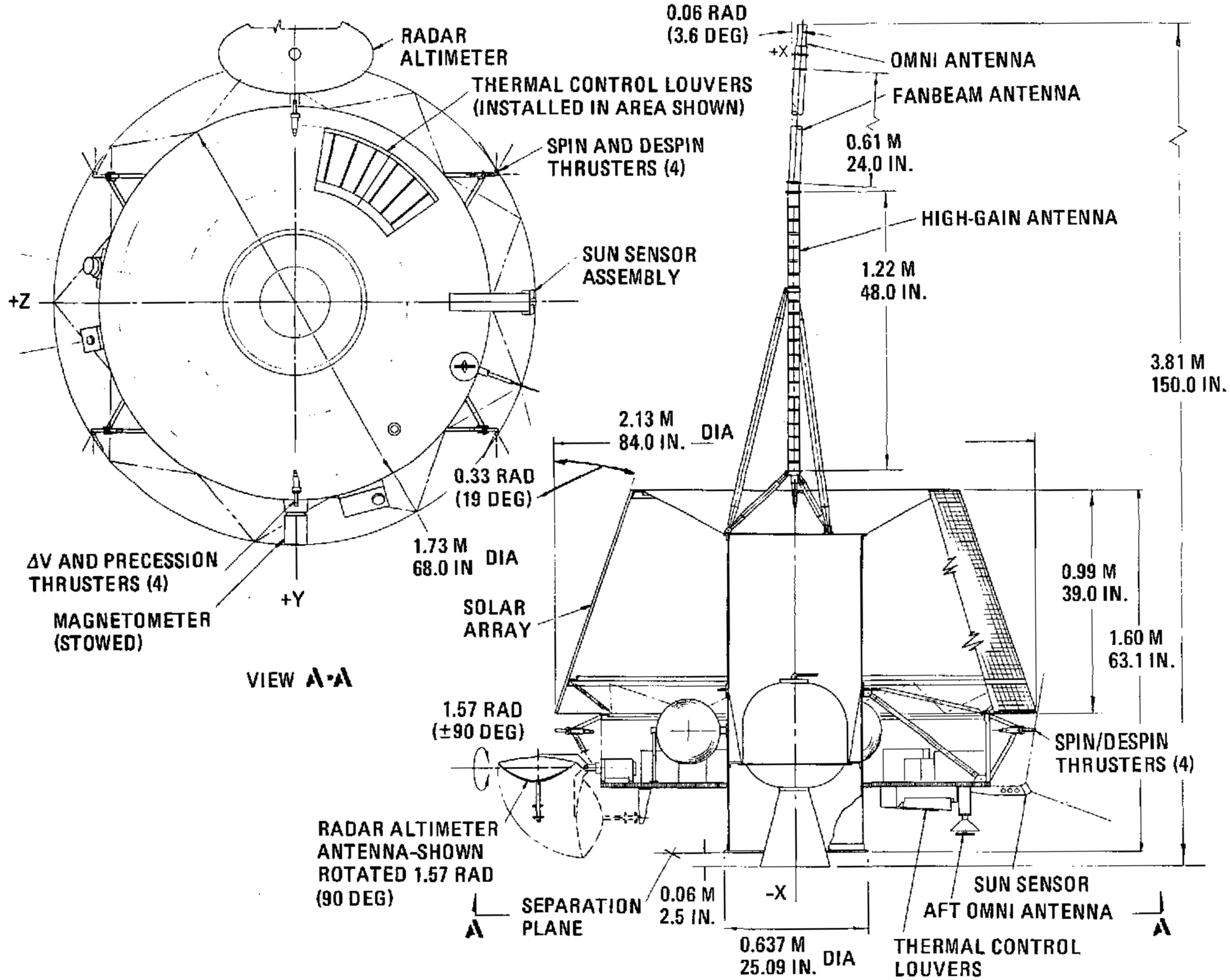


Figure 5-29. Thor/Delta Orbiter 36-Watt Fanbeam/Fanscan

31 W T/D III

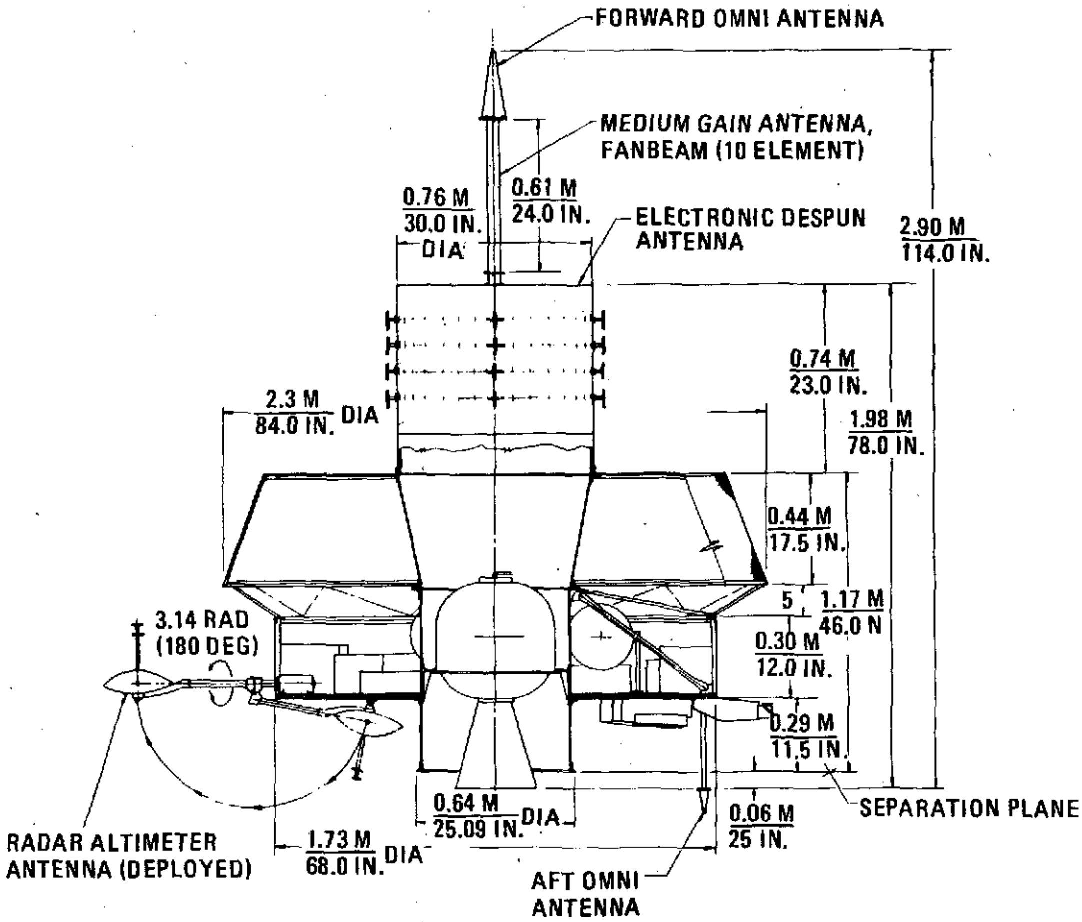


Figure 5-30. Thor/Delta Orbiter Mission Spacecraft Normal-to-Venus Orbital Plane Electronic Despun Antenna

neutral mass spectrometers; its design would use the same deployment/ gimballing device used by the radar altimeter antenna experiment. This configuration requires software programming and at least weekly updating to maintain ram pointing angles, which adds some burden to flight operations activities.

Table 5-6 compares the weights of the four Thor/Delta orbiter configurations, and includes, for the preferred configuration, the changes required to meet a minimum acceptable weight contingency. It is noted that for the Atlas/Centaur orbiter configurations the weight saving (and costly) changes are not required, as will be seen in the next paragraph.

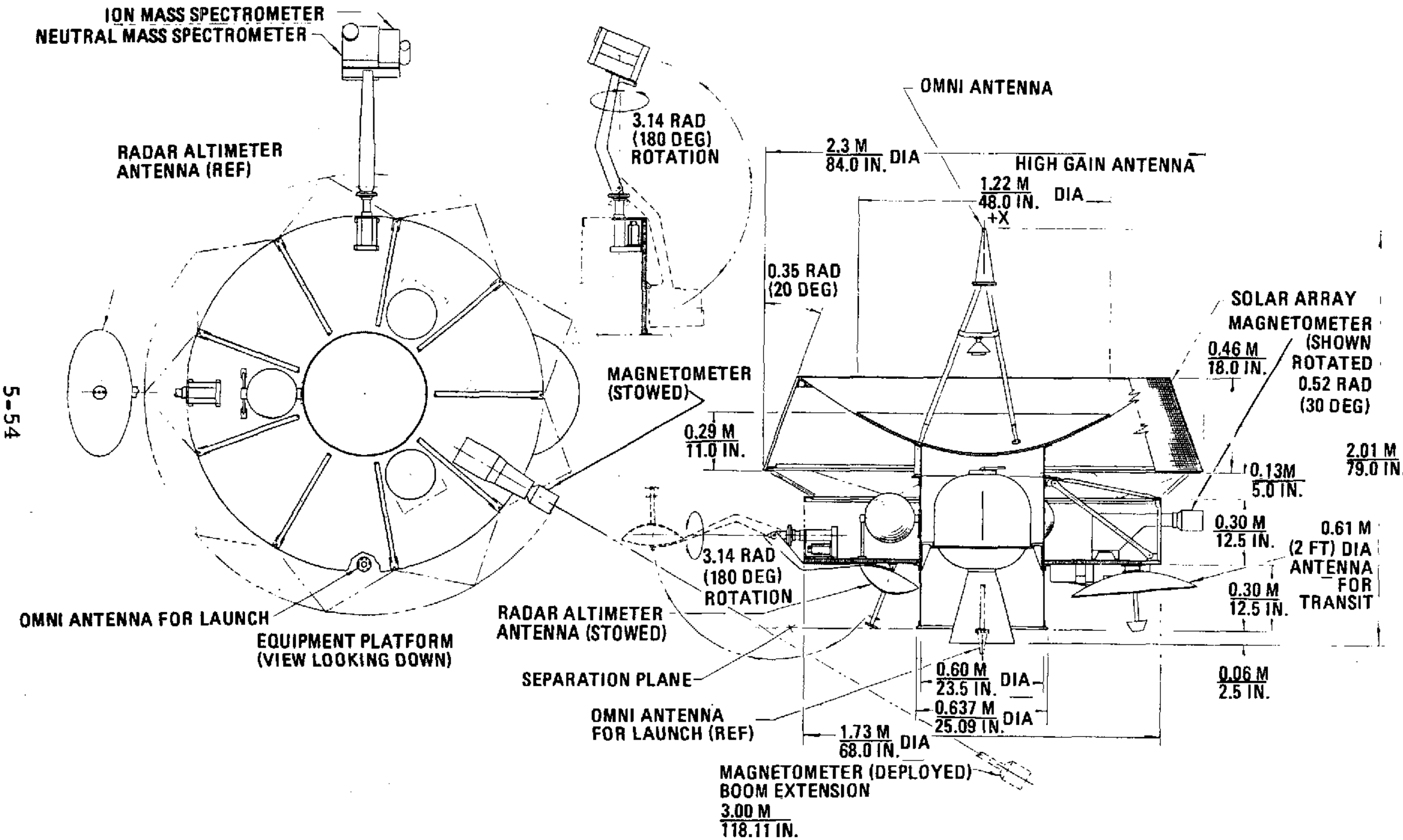


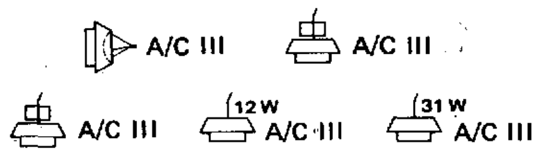
Figure 5-31. Thor/Delta Earth Pointer

Table 5-6. Thor/Delta Orbiter Configuration Weight Comparison Summary

DESCRIPTION	31-WATT FANBEAM FANSCAN [KG (LB)]		12-WATT FANBEAM FANSCAN [KG (LB)]		EARTH POINTER [KG (LB)]		12-WATT DESPUN REFLECTOR [KG (LB)]	
ELECTRICAL POWER	42.1	(92.8)	30.4	(66.9)	30.4	(66.9)	34.7	(76.4)
COMMUNICATIONS	15.9	(35.0)	11.5	(25.4)	15.4	(34.0)	12.4	(27.4)
ELECTRICAL DISTRIBUTION	12.6	(27.8)	12.6	(27.8)	12.6	(27.8)	12.6	(27.8)
DATA HANDLING	5.7	(12.5)	5.7	(12.5)	5.7	(12.5)	5.7	(12.5)
ATTITUDE CONTROL	2.3	(5.1)	2.3	(5.1)	2.3	(5.1)	12.4	(27.4)
PROPULSION (DRY)	6.8	(14.9)	6.8	(14.9)	6.8	(14.9)	6.8	(14.9)
SOLID INSERTION MOTOR (BURNOUT)	9.1	(20.0)	9.1	(20.0)	9.1	(20.0)	9.1	(20.0)
THERMAL CONTROL	14.9	(32.9)	11.2	(24.8)	11.2	(24.8)	11.2	(24.8)
STRUCTURE	40.4	(89.0)	41.9	(92.3)	44.2	(97.4)	40.9	(90.1)
BALANCE WEIGHT PROVISION	2.7	(6.0)	2.7	(6.0)	2.7	(6.0)	2.7	(6.0)
SPACECRAFT BUS LESS SCIENCE (DRY)	152.5	(336.0)	134.2	(295.7)	140.4	(309.4)	148.5	(327.3)
SCIENTIFIC INSTRUMENTS	28.3	(62.5)	28.3	(62.5)	28.3	(62.5)	28.3	(62.5)
SPACECRAFT (DRY)	180.8	(398.5)	162.5	(358.2)	168.7	(371.9)	176.8	(389.8)
INSERTION MOTOR EXPENDABLES	84.4	(186.1)	84.4	(186.1)	84.4	(186.1)	84.4	(186.1)
HYDRAZINE PROPELLANT AND PRESSURANT	17.4	(38.3)	17.4	(38.3)	17.4	(38.3)	17.4	(38.3)
SPACECRAFT LESS CONTINGENCY	282.6	(622.9)	264.3	(582.6)	270.5	(596.3)	278.6	(614.2)
CONTINGENCY (NET ALLOWABLE)*	10.0 (5.5%)	(22.1)	28.3 (17.4%)	(62.4)	22.1 (13.1%)	(48.7)	14.0 (7.9%)	(30.8)
GROSS SPACECRAFT AFTER SEPARATION	292.6	(645.0)	292.6	(645.0)	292.6	(645.0)	292.6	(645.0)

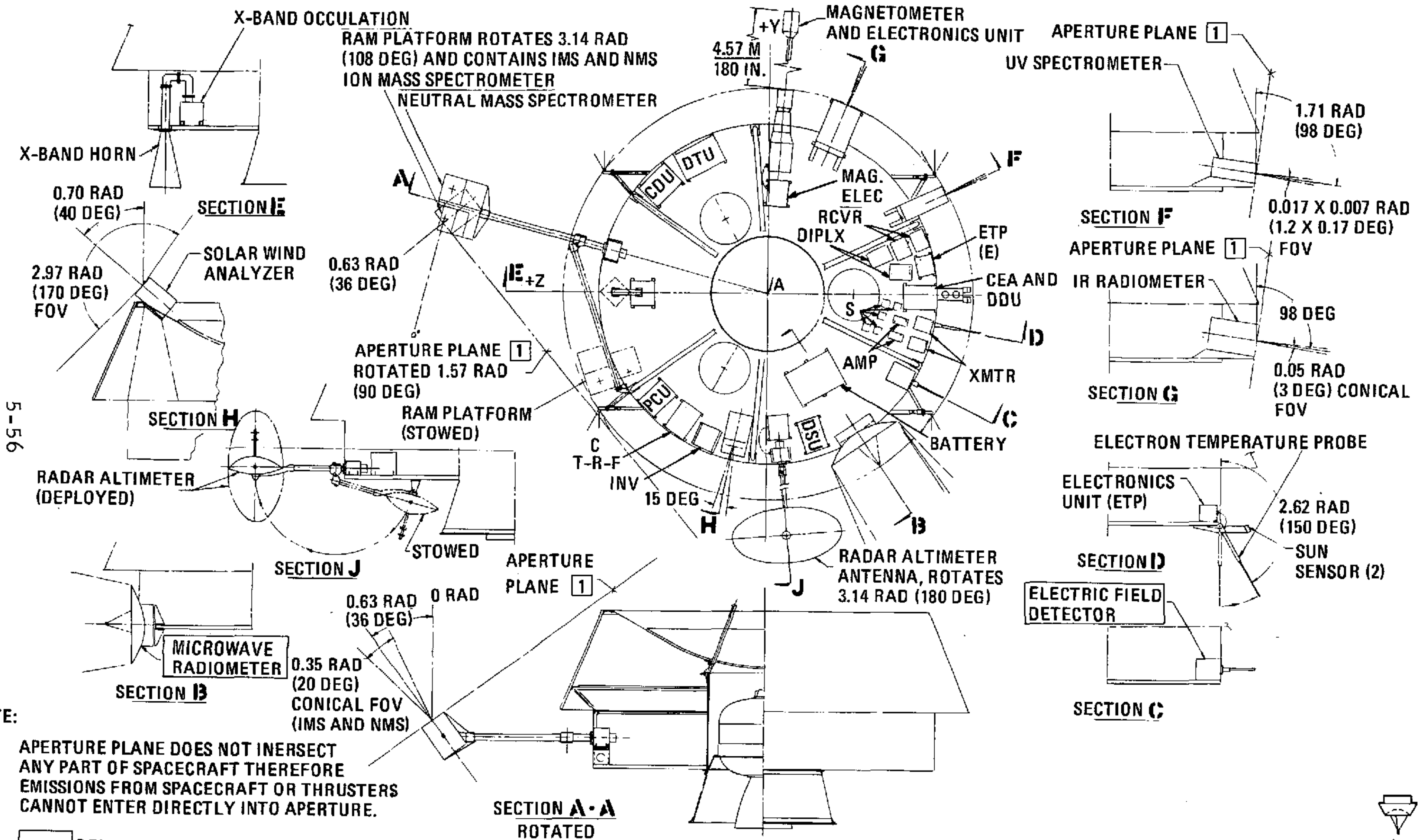
*PERCENTAGE VALUES ARE RELATIVE TO THE DRY SPACECRAFT WEIGHT.

5.2.5.2 Atlas/Centaur Configurations



The Atlas/Centaur orbiter configurations are based on maximizing commonality with the Atlas/Centaur probe bus, which in turn was scaled up from the Thor/Delta bus configuration to accommodate the larger, more cost-effective Atlas/Centaur probes. Thus, the Atlas/Centaur orbiters are similar to the Thor/Delta orbiters but scaled up to be compatible with the larger Centaur shroud.

Figures 5-32 through 5-35 show the preferred Atlas/Centaur configuration and the designs for Options 1, 2, and 3 (which correspond to the Thor/Delta designs). The main difference between the Atlas/Centaur options and the Thor/Delta versions is that the Atlas/Centaur options do not incorporate the weight-saving items that were necessary to meet the Thor/Delta weight constraints. Table 5-7 is a weight comparison summary of the four Atlas/Centaur orbiters and shows that, in each case, the available contingency is more than sufficient.



NOTE:

1 APERTURE PLANE DOES NOT INERSECT ANY PART OF SPACECRAFT THEREFORE EMISSIONS FROM SPACECRAFT OR THRUSTERS CANNOT ENTER DIRECTLY INTO APERTURE.

2. [ ] DENOTES OTHER CANDIDATE INSTRUMENTS.

Figure 5-32. Preferred Atlas/Centaur Orbiter Mission Earth-Pointing Spacecraft, Version IV Science Payload

12 W
A/C III

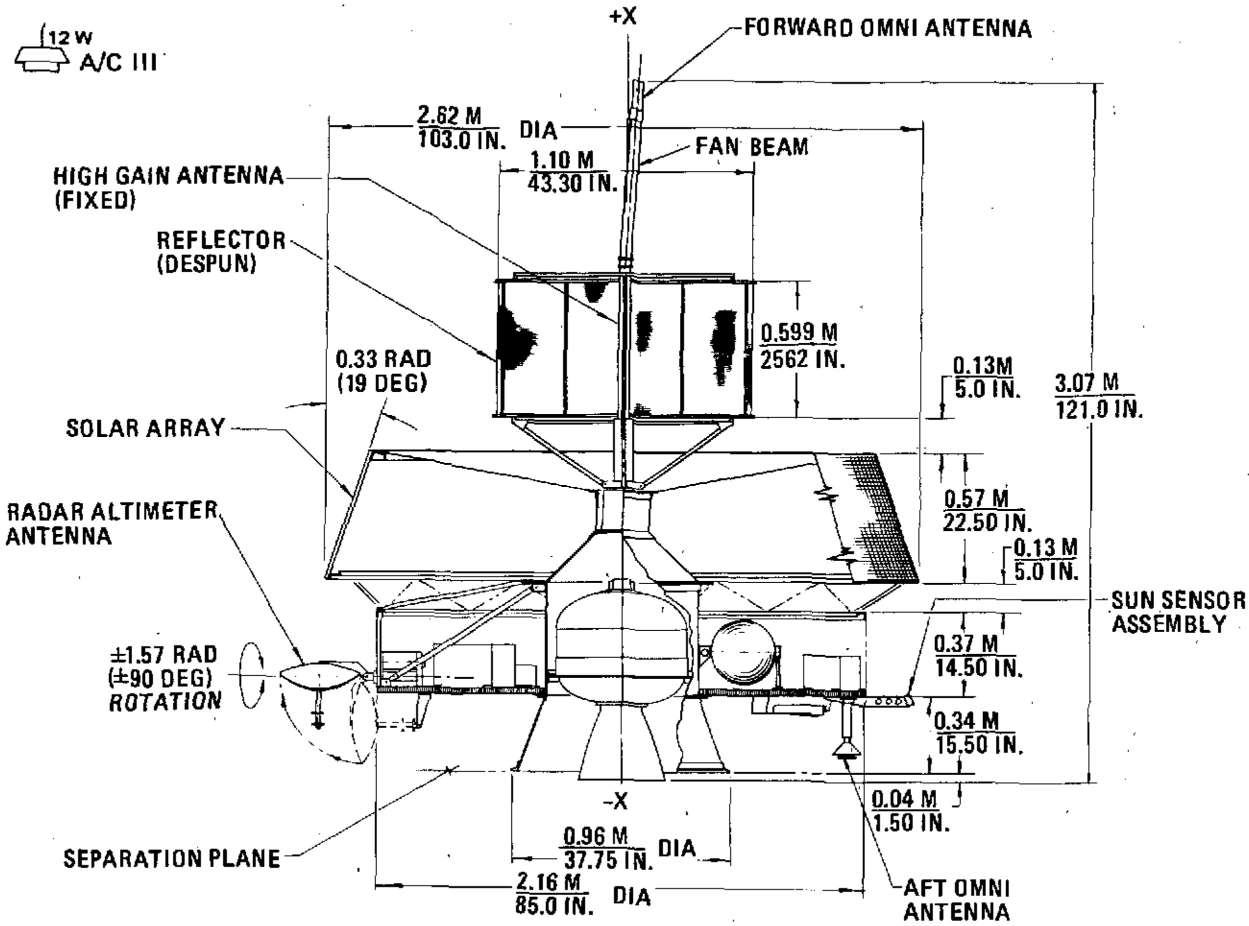


Figure 5-33. Atlas/Centaur Orbiter Option 2 Despun Antenna

Table 5-7. Atlas/Centaur Orbiter Configuration Weight Comparison Summary

DESCRIPTION	OPTION 2 31-WATT FANBEAM, FANSCAN [KG (LB)]		OPTION 3 12-WATT FANBEAM, FANSCAN [KG (LB)]		PREFERRED EARTH-POINTER [KG (LB)]		OPTION 1 12-WATT DESPUN REFLECTOR [KG (LB)]	
ELECTRICAL POWER	50.8	(112.0)	35.7	(78.6)	39.4	(86.8)	44.5	(98.1)
COMMUNICATIONS	12.5	(27.5)	11.8	(25.9)	15.0	(33.1)	12.6	(27.9)
ELECTRICAL DISTRIBUTION	15.8	(34.8)	15.8	(34.8)	15.8	(34.8)	15.8	(34.8)
DATA HANDLING	12.5	(27.5)	12.5	(27.5)	18.4	(40.5)	18.4	(40.5)
ATTITUDE CONTROL	2.7	(6.0)	2.7	(6.0)	4.7	(10.3)	12.8	(28.3)
PROPULSION (DRY)	6.9	(15.3)	6.9	(15.3)	7.9	(17.4)	7.9	(17.4)
SOLID INSERTION MOTOR (BURNOUT)	18.7	(41.1)	18.7	(41.1)	18.7	(41.1)	18.7	(41.1)
THERMAL CONTROL	20.2	(44.5)	15.0	(33.0)	15.2	(33.5)	15.0	(33.0)
STRUCTURE	73.4	(161.8)	73.0	(161.1)	73.7	(162.5)	72.0	(158.8)
BALANCE WEIGHT PROVISION	5.4	(12.0)	5.4	(12.0)	5.4	(12.0)	5.4	(12.0)
SPACECRAFT BUS LESS SCIENCE (DRY)	218.9	(482.5)	197.5	(435.3)	214.1	(472.0)	223.1	(491.9)
SCIENTIFIC INSTRUMENTS	33.0	(72.9)	33.0	(72.9)	45.4	(100.1)	45.4	(100.1)
SPACECRAFT (DRY)	251.9	(555.4)	230.5	(508.2)	259.5	(572.1)	268.5	(592.0)
INSERTION MOTOR EXPENDABLES	126.1	(278.0)	126.1	(278.0)	144.5	(318.5)	144.5	(318.5)
HYDRAZINE PROPELLANT AND PRESSURANT	14.2	(31.4)	14.2	(31.4)	18.1	(39.9)	18.1	(39.9)
SPACECRAFT LESS CONTINGENCY	392.2	(864.8)	370.8	(817.6)	422.1	(930.5)	431.1	(950.4)
CONTINGENCY (NET ALLOWABLE)*	43.2 (17.1%)	(95.2)	64.6 (28.0%)	(142.4)	85.9 (33.1%)	(189.5)	76.9 (28.6%)	(169.6)
GROSS SPACECRAFT AFTER SEPARATION	435.4	(960.0)	435.4	(960.0)	508.0	(1120.0)	508.0	(1120.0)

*PERCENTAGE VALUES ARE RELATIVE TO THE DRY SPACECRAFT WEIGHT.

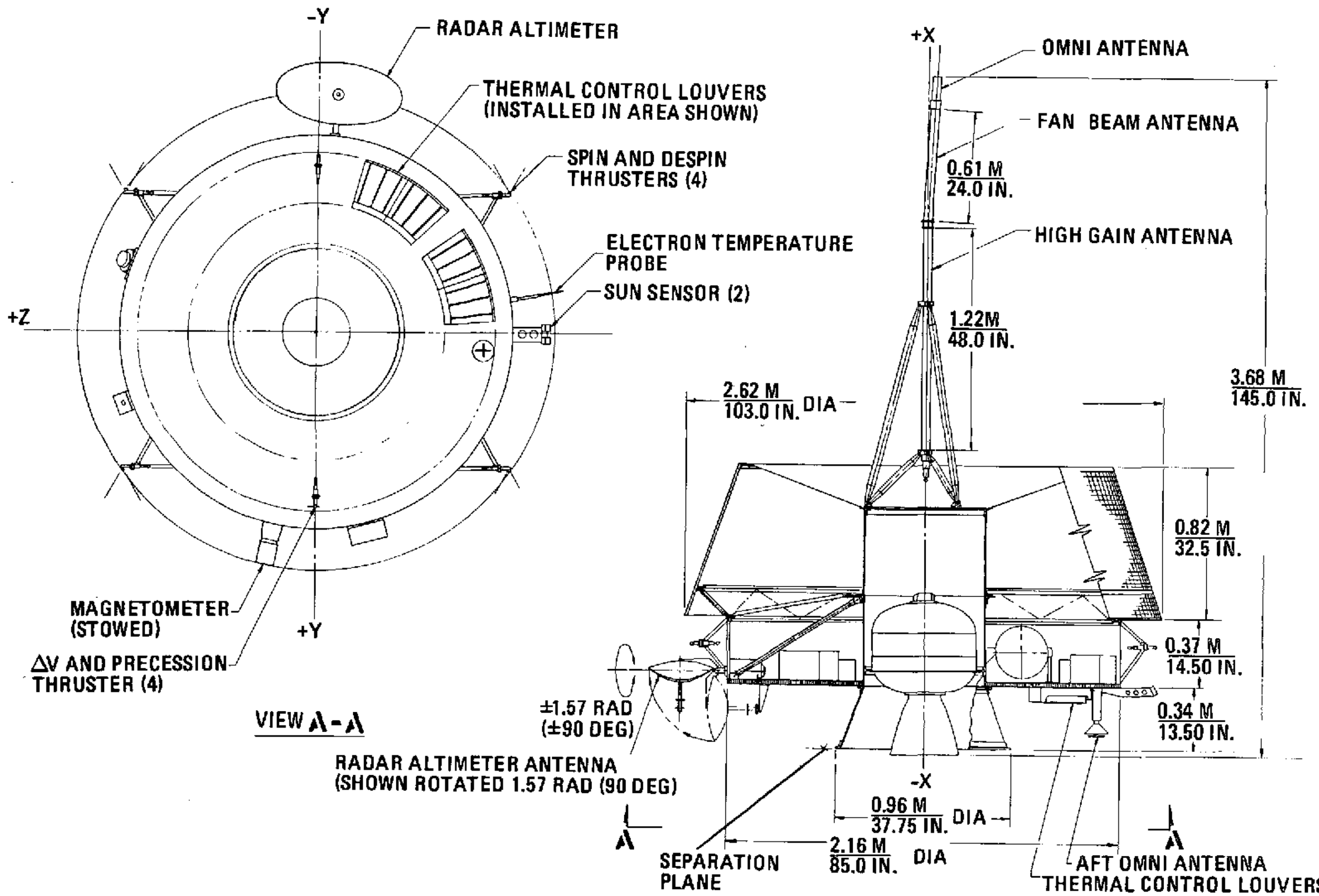


Figure 5-34. Atlas/Centaur Orbiter Option 2, 31-Watt Fanbeam/Fanscan

12 W A/C III

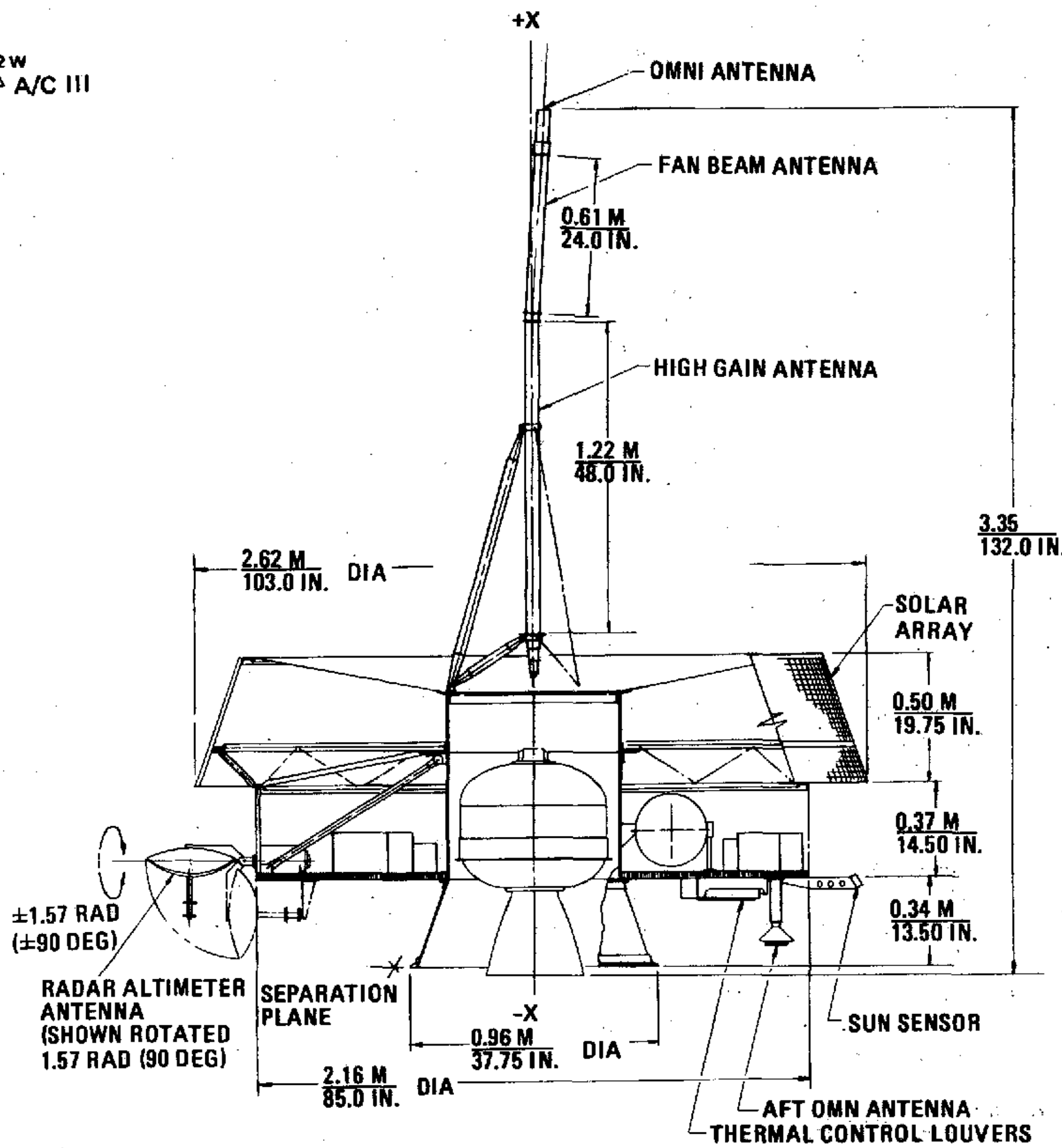


Figure 5-35. Atlas/Centaur Orbiter Option 3, 12-Watt Fanbeam/Fanscan

For the Version IV science payload, only the preferred and Option 1 orbiters are viable, as shown in Figures 5-16 and 5-17, because of the limited data rate capabilities of the other configurations.

5.2.5.3 Preferred Atlas/Centaur Orbiter Configuration A/C IV

The preferred orbiter configuration is shown in Figure 5-36. It is the lowest cost design that meets the requirements of the Version IV payload and all 1978-launches with Atlas/Centaur.

For the Version III science payload, specifying the Atlas/Centaur for both missions, we had selected the Thor/Delta 31-watt fanbeam fan-scan configuration as the lowest cost design, which could operate with the 26 meter DSN subnet. Our studies had shown that the additional cost of the Atlas/Centaur launch vehicle was greater than the additional cost of designing and building an orbiter for for the Thor/Delta (which would meet the original program requirements and have an adequate weight contingency for the Thor/Delta).

NASA's selection of the Atlas/Centaur for both missions, therefore, implies that additional cost factors are being considered (beyond the $9-million launch vehicle cost that was originally provided as a working figure). The following items summarize our understanding of these additional factors:

- Savings arising from the use of a common launch vehicle for two launches 3 months apart. These include launch vehicle procurement, NASA management costs and reduced launch operations cost.
- Uncertainty in the definition of the orbiter science instruments and their requirements, and lack of margin in the Thor/Delta orbiter to meet possible increased requirements.
- The desire to avoid the development of a spacecraft which is too constrained to be useful in possible follow-on missions to Venus or Mars.

5.2.6 ESRO Configurations

The primary impact of ESRO participation is the orbit insertion system and antenna for the orbiter mission.

ESRO's preferred orbit insertion system is the regulated pressure-fed liquid bipropellant Symphonie retro design, as described in the MBB study. It is fully qualified. Figure 5-37 shows the orbiter spacecraft with this N_2O_4/Aerozine 50 monotank system and the Helios despun reflector installed as sized to be launched by the Thor/Delta. The main impact on the configuration is the size of the central cylinder required to accommodate the propellant tank. The despun reflector of the Option 1 orbiter (see Section 5.2.4) is from Helios and is the antenna recommended to ESRO in the MBB study. If this is the selected system for the orbiter

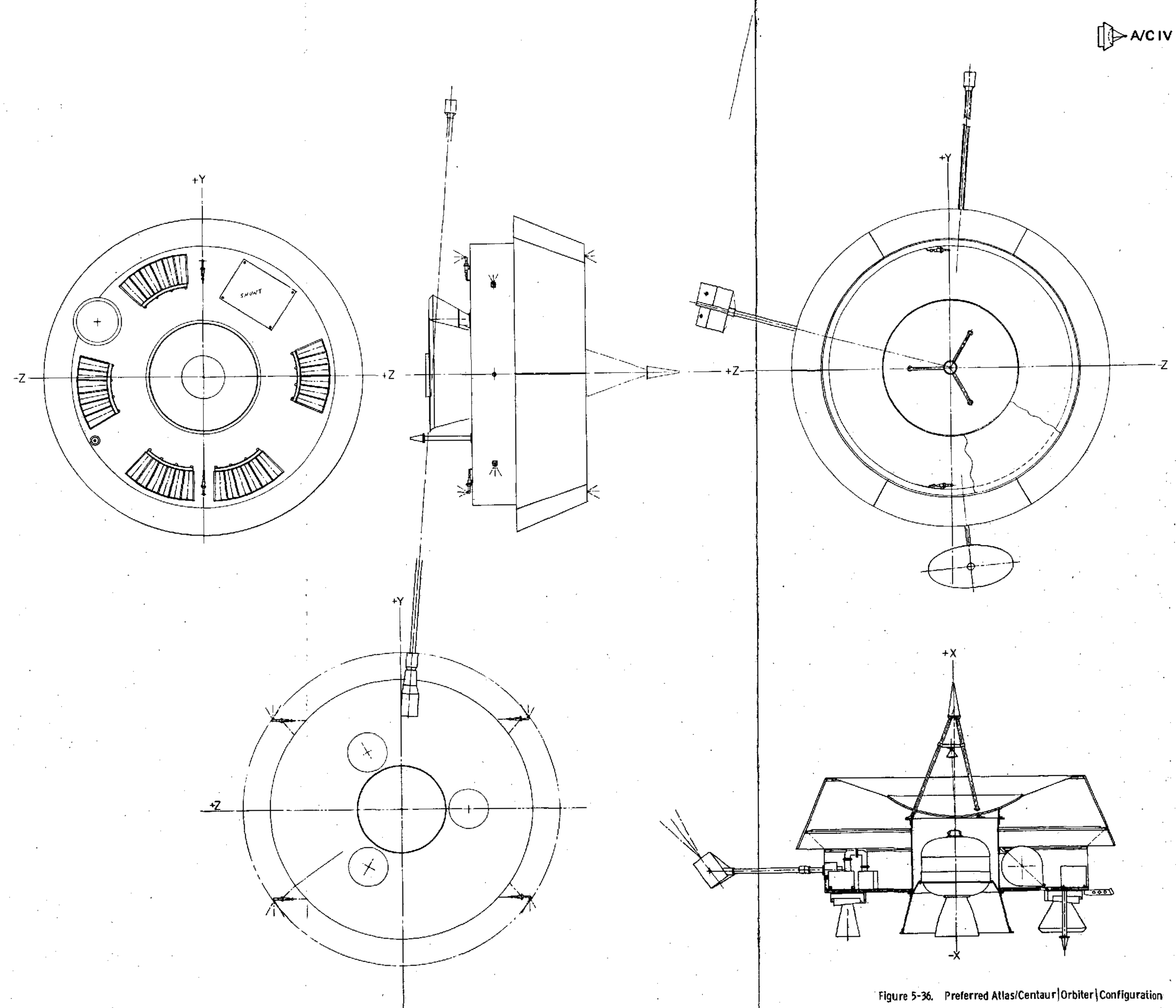


Figure 5-36. Preferred Atlas/Centaur|Orbiter|Configuration

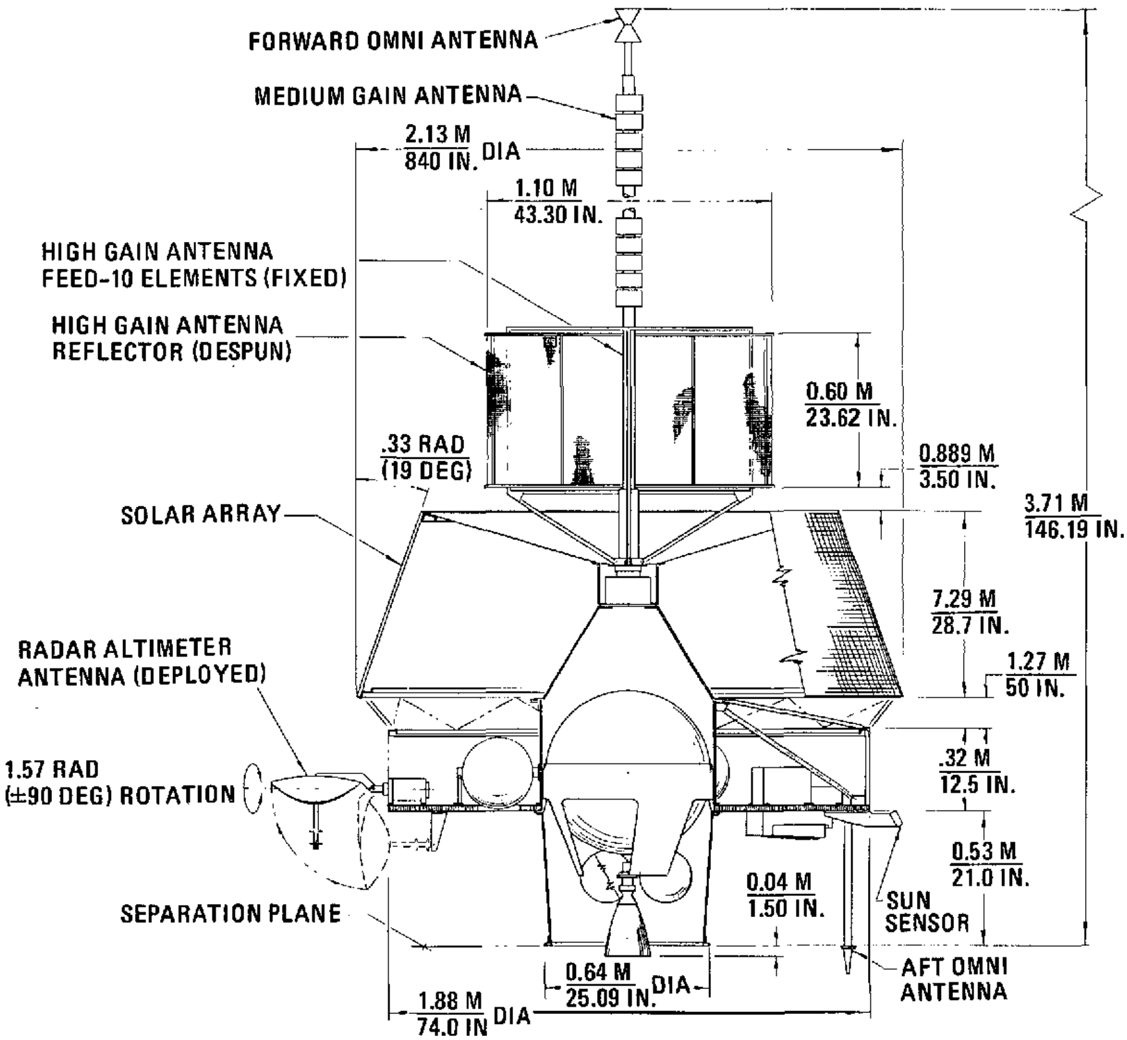


Figure 5-37. Thor/Delta Mission Spacecraft ESRO Propulsion and Antenna

spacecraft, the probe bus spacecraft will be configured with a corresponding central cylinder, for maximum structural commonality between missions, as shown in Figure 5-38.

5.2.6.1 Liquid Versus Solid Orbit Insertion Propulsion

An alternative to the liquid retro system is the SIRIO solid propellant motor, as described in the MBB study. This motor would be interchangeable with the all-NASA solid orbit insertion motor (described in Section 8.6), and therefore would have no impact on the configuration.

When used with the Thor/Delta, the multiple restart capability of the liquid engine opens up the possibility of also using it as a fourth stage

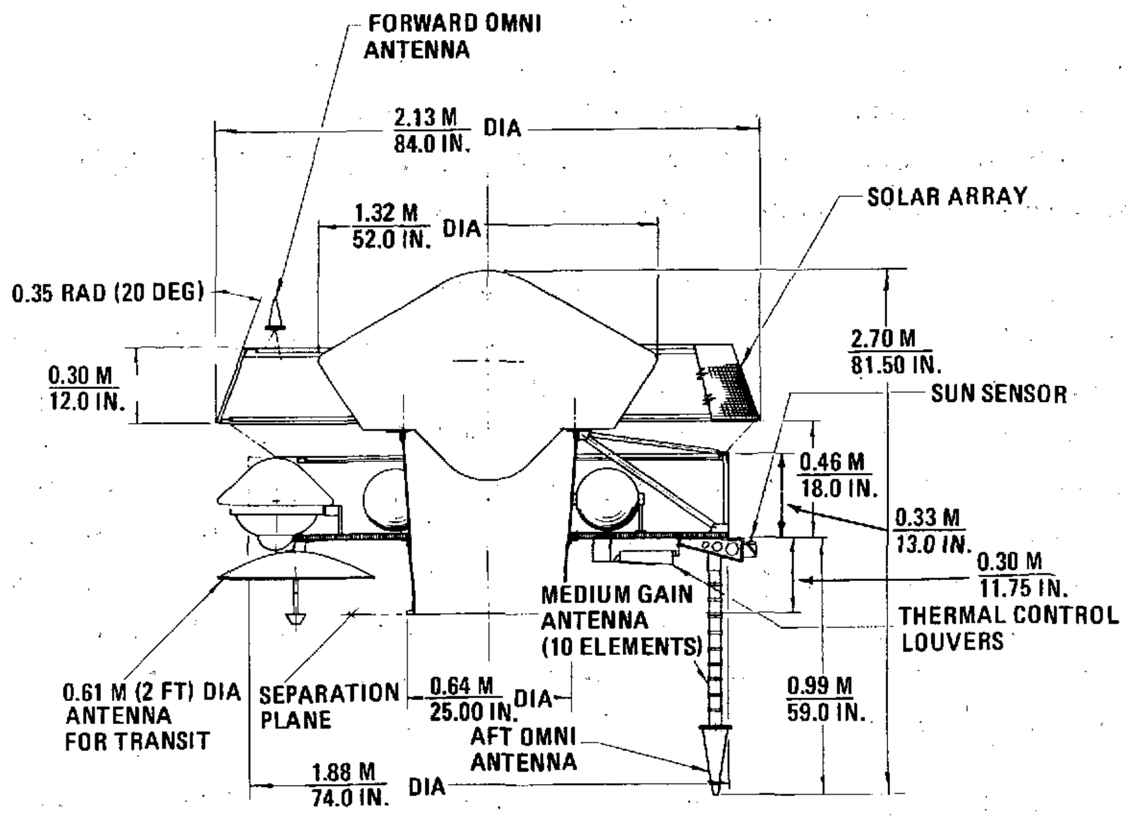


Figure 5-38. Thor/Delta Probe Mission Spacecraft ESRO Propulsion

and to perform the first midcourse correction. It appears that a net payload increase to venus orbit of as much as 13.6 kilograms (30 pounds) could be realized with this multiple use; if used only for first midcourse correction the increase would be 2.5 kilograms (5.5 pounds), which results from the difference between the specific impulse of the bipropellant (305 seconds) versus the hydrazine RCS (220 seconds). In either case additional pyrotechnique valves would be required to insure no leakage during the cruise period. In the Atlas/Centaur version, the full capability of the tankage is devoted to orbit insertion. There is no known European solid propellant motor suitable for this version.

5.2.7 Spacecraft/Launch Vehicle Mechanical Interfaces

The envelope describing the maximum physical dimensions of the spacecraft is shown in Figures 5-39 and 5-40 for Thor/Delta and Atlas/Centaur launch vehicles.

For Thor/Delta, the interface with the TE-364 third stage is a standard 25 x 12 or 37 x 31 spacecraft-to-launch-vehicle adapter; the former is structurally adequate and significantly lighter than the latter and so is the recommended selection. Spinup is accomplished by a standard spin table, before firing the third stage motor.

The interface with the Centaur stage is a standard Mariner-Venus-Mercury (MVM) adapter, to which a new spacecraft attach fitting will be bolted. This new fitting interfaces with the spacecraft, using a separable V-band clamp. An existing, ordnance-actuated 0.96 meter (37.75 inch) diameter V-band clamp will be used for spacecraft separation from the third stage. This is shown in Figure 5-40. Thus, a new attach fitting will have to be developed, but the critical V-band clamp will be an existing design. No spin table is required; the Centaur stage will provide substantially all of the required spinup, with the spacecraft thrusters supplying any remainder to attain the desired rate.

An alternative for Atlas/Centaur may be to use the MVM adapter with the MVM attach fitting, as described in Appendix D to NASA/Ames Specification No. 2-17502. The spacecraft would mate to the attach fitting by means of an existing 1.4 meter (55-inch) diameter, ordnance activated V-band clamp. This scheme would impose a requirement on the spacecraft to mate at the 1.4 neter (55 inch) diameter. The mating spacecraft aft skirt would have to be of truss construction, to enable the thermal louvers on the aft surface of the spacecraft equipment platform to have a minimum of reflection from the skirt. Since it is desirable to retain the 0.64 meter (25 inch) diameter central cylinder to accommodate the orbit insertion motor (and support the large probe on the probe bus) the aft skirt would be relatively shallow which, in turn, would impose high radial loads on the eight mating projections of the MVM attach fitting. These loads are not consistent with the loads imposed on the fitting by the MVM spacecraft, and therefore the fitting would require extensive design. Thus, a new attach fitting is a more effective choice: the spacecraft skirt is not complicated, and load paths are smooth.

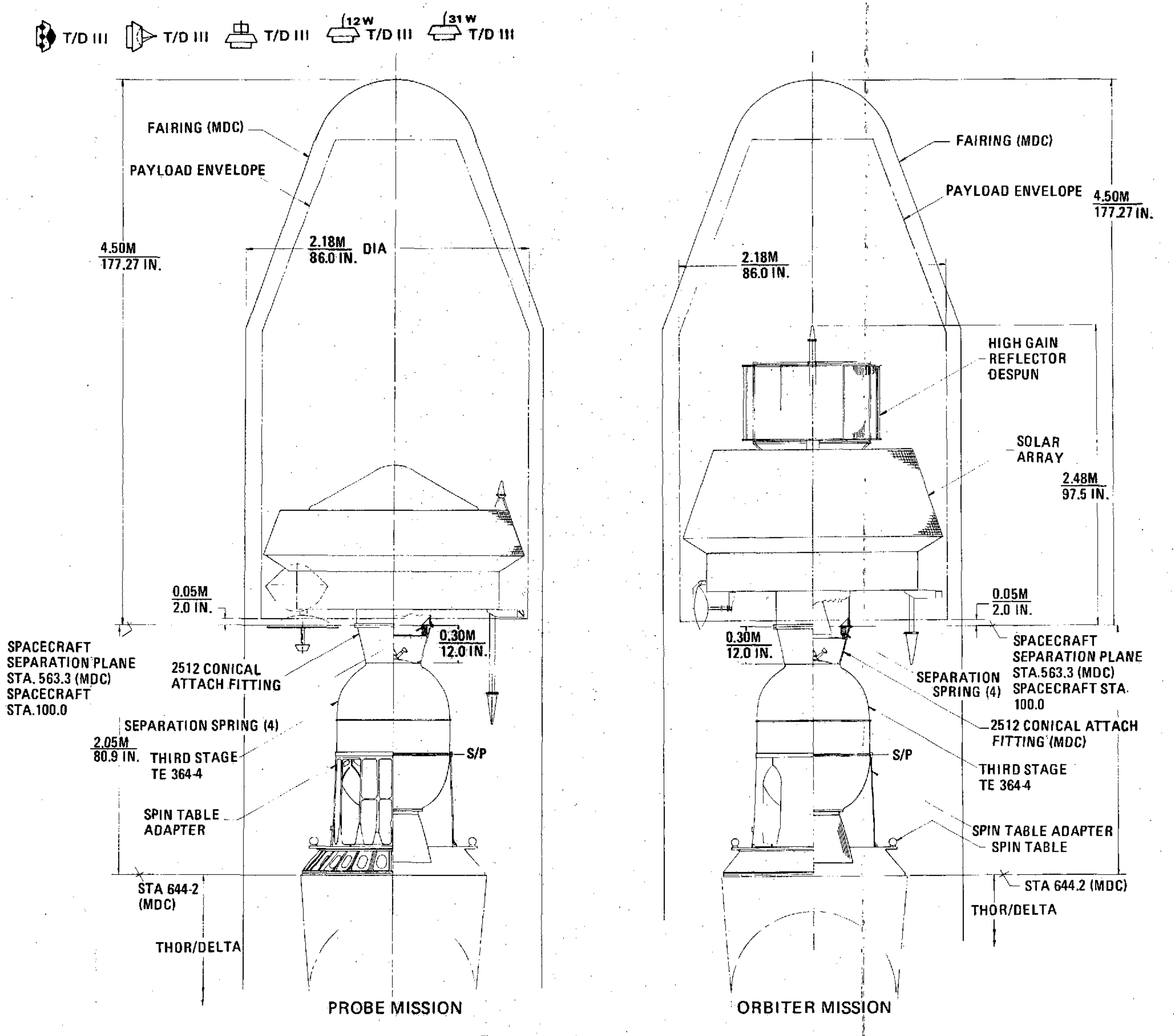


Figure 5-39. Thor/Delta Launch Configuration

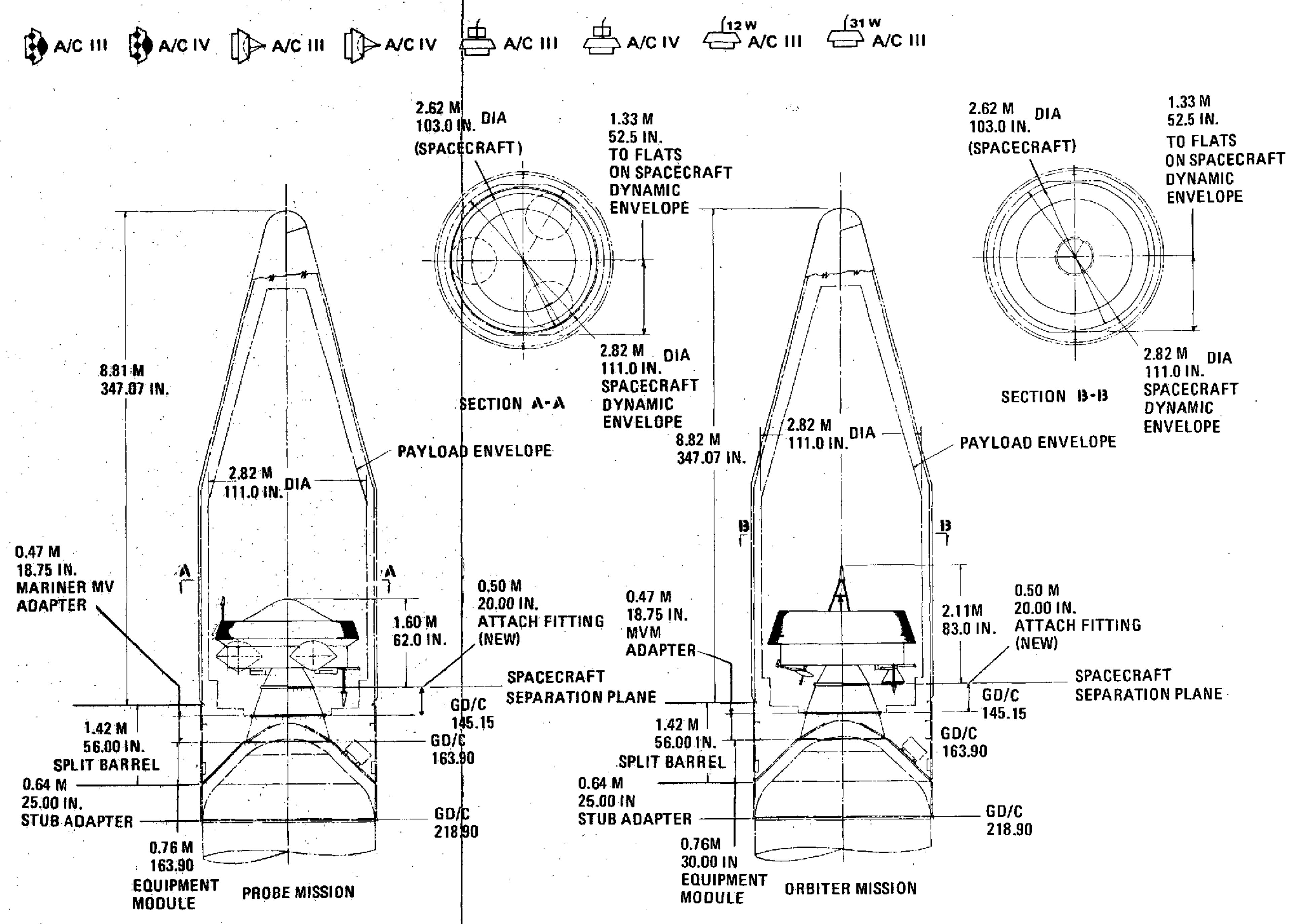


Figure 5-40. Atlas/Centaur Launch Configuration

5.2.8 Preliminary Design Loads and Environments

Based on flight experience with both Thor/Delta and Atlas/Centaur booster system, preliminary design loads and environments have been tabulated in Tables 5-8 through 5-17, and Figure 5-41. These include recommended component qualification levels. Aerodynamic response analysis to reflect the actual response of the spacecraft-booster stackup has been performed to update these preliminary values; this is described in Section 8.8. During the hardware design phase, it will be repeated to give the final design and test environments.

Table 5-8.
Thor/Delta Design Loads (Limit)
(Ultimate Factor = 1.5)

CONDITION	AXIAL (G)	LATERAL (G)
LIFTOFF	+2.9, -1.0	±2.3
POGO	+9.7, -0.3	±0.5
MECO	+12.0, -0	±0.65
THIRD STAGE BURNOUT	+16.0	COMBINE AXIAL WITH 9.42 RAD/S (90 RPM) SPIN RATE (0.23 G/IN)
ORBIT INSERTION (ORBITER ONLY)	+16.0	COMBINE AXIAL WITH 6.28 RAD/S (60 RPM) SPIN RATE
STIFFNESS REQUIRED	35 HZ	20 HZ

Table 5-9.
Thor/Delta Acoustic Levels
(Qualification Level)

FREQUENCY	SOUND PRESSURE LEVEL (DB)
37.5 TO 75	130
75 TO 150	135
150 TO 300	138
300 TO 600	140
1200 TO 2400	138
2400 TO 4800	134
4800 TO 9600	129
OVERALL	146
DURATION	1 MIN

Table 5-10. Thor/Delta Vibration Environments
(Qualification Level)

TYPE	DIRECTION	FREQUENCY (HZ)	ACCELERATION (G)
SINUSOIDAL	AXIAL	5 TO 15	2.3 G OR 0.013 M (0.5 IN) DOUBLE AMPLITUDE
		15 TO 21	6.8
		21 TO 100	2.3
	LATERAL	5 TO 14	2.3 G OR 0.012 M (0.5 IN) DOUBLE AMPLITUDE)
		14 TO 100	1.5

TYPICAL – RANDOM VIBRATION

DIRECTION	FREQUENCY (HZ)	PSD LEVEL (G^2/HZ)	ACCELERATION [G (RMS)]	DURATION (MIN/AXIS)
THREE MUTUALLY ⊥AXES	20 TO 300	+3 DB/OCT	--	--
	300 TO 2000	0.045	9.2	2

Table 5-11.
Atlas/Centaur Design Loads (Limit) (UltimateFactor = 1.5)

CONDITION	AXIAL (G)	LATERAL (G)
LAUNCH	6.5	±2.0
ORBIT INSERTION	8.0	COMBINE AXIAL WITH 6.28 RAD/S (60 RPM) SPIN RATE

Table 5-12.
Atlas/Centaur Vibration Environments (Qualification Level)

TYPE	DIRECTION	FREQUENCY (HZ)	ACCELERATION (G)
SINUSOIDAL	AXIAL	5 TO 8.5	0.015 M (0.6 IN) DOUBLE AMPLITUDE (2 OCT/MIN)
		8.5 TO 2000	±2.3 G (2 OCT/MIN)
	LATERAL	5 TO 8	0.011 M (0.45 IN) DOUBLE AMPLITUDE
		8 TO 2000	±1.5G

TYPICAL – RANDOM VIBRATION

DIRECTION	FREQUENCY (HZ)	PSD LEVEL (G^2/HZ)	ACCELERATION [G (RMS)]	DURATION (MIN/AXIS)
THREE MUTUALLY ± AXES	20 TO 150	6 DB/OCT	—	—
	150 TO 2000	0.045	9.3	4

Table 5-13.
Atlas/Centaur Acoustic Levels (Qualification Level)

OCTAVE BAND CENTER FREQUENCY (HZ)	SOUND PRESSURE LEVEL (DB)
15.8	119
31.5	128
63	134
125	138
250	141
500	138
1000	132
2000	132
4000	132
8000	132
OVERALL	146
DURATION	2 MIN

Table 5-14.
Component Design Criteria Sustained Acceleration (Qualification Level)

DIRECTION	ACCELERATION (G)
AXIAL	20
LATERAL (RADIAL)	10

Table 5-15.
Component Sinusoidal Vibration (Qualification Level)

DIRECTION	FREQUENCY (HZ)	ACCELERATION (G 0-P)
AXIAL	5 TO 15	5 G OR 0.013 M (0.5 IN) DOUBLE AMPLITUDE
	15 TO 21	15
	21 TO 35	7.5
	35 TO 50	5.0
	50 TO 100	3.0
LATERAL	5 TO 30	5 G OR 0.013 M (0.5 IN) DOUBLE AMPLITUDE
	30 TO 100	3.0

Table 5-16. Component Random Vibration (Qualification Level)

DIRECTION	FREQUENCY RANGE (HZ)	PSD LEVEL (G^2/HZ)	ACCELERATION [G (RMS)]	DURATION (MIN/AXIS)
THREE MUTUALLY ⊥ AXES	20 TO 60	0.05	19.6	1
	60 TO 300	+3 DB/OCT		
	300 TO 1200	0.25		
	1200 TO 2000	-6 DB/OCT		

Table 5-17. Component Acoustic Levels (Qualification Level)

OCTAVE BAND CENTER FREQUENCY (HZ)	SOUND PRESSURE LEVEL (DB)
15.8	119
31.5	128
63	134
125	138
250	141
500	138
1000	132
2000	132
4000	132
OVERALL	146

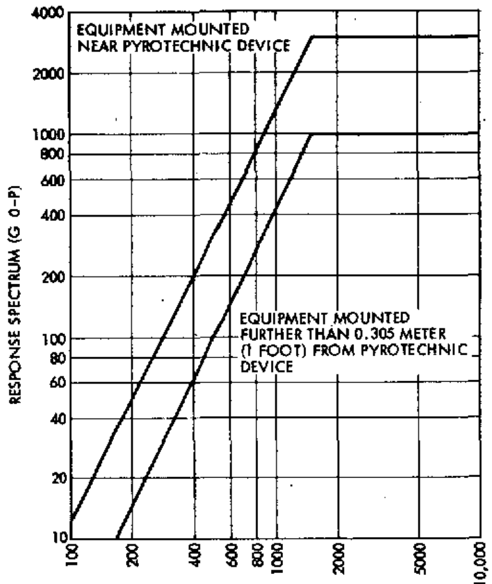


Figure 5-41. Typical Response Spectra Envelopes for Spacecraft Pyrotechnic Release Shock (5 Percent Damping)

6. Spacecraft System Definition

C-2

6.1 Probe System Definition

6. SPACECRAFT SYSTEM DEFINITION

6.1 PROBE SYSTEM DEFINITION

The probe system design requirements for the Thor/Delta (T/D) and Atlas/Centaur (A/C) configurations were established by the science objectives and requirements. The significant probe design requirements used for this study are summarized in Section 5.1. The preferred Atlas/Centaur and Thor/Delta probe system configurations are discussed in detail in subsequent paragraphs.

6.1.1 Atlas/Centaur Probes

The main objective in the A/C probe system configuration selection was to attain minimum cost. The increased weight and volume were used in the most advantageous manner to meet that objective. Definitions of the resulting designs and their performance follow.

6.1.1.1 Aerodynamic and Flight Dynamic Performance

The aerodynamic configurations of the preferred probes are shown in Figure 6-1. These shapes have been established through studies based on existing MMC data (Viking), data in the literature, and tests carried out under the MMC IRAD program in parallel with the present study. Three different shapes are identified because each configuration calls for a different ballistic coefficient, must function through a different Mach number range, and has different mechanical requirements. The large probe entry vehicle configuration functions from hypersonic entry to Mach number 0.7 where the parachute is deployed and the nose cone separated. Subsequently, the descent capsule is released and it must function under subsonic, terminal flight conditions until it impacts the surface of Venus. The small probe must function throughout the entire Mach number range from hypersonic to subsonic. The aerodynamic coefficients and derivatives for the three configurations are presented in Section 7.1.

An extremely important aspect of the probe requirements is that they must exhibit completely passive dynamic stability so as to provide stable platforms for the science instruments. Because the probes are spin stabilized during the preentry coast period, care must be taken to avoid roll resonance effects during the entry. Also, because the probes transmit

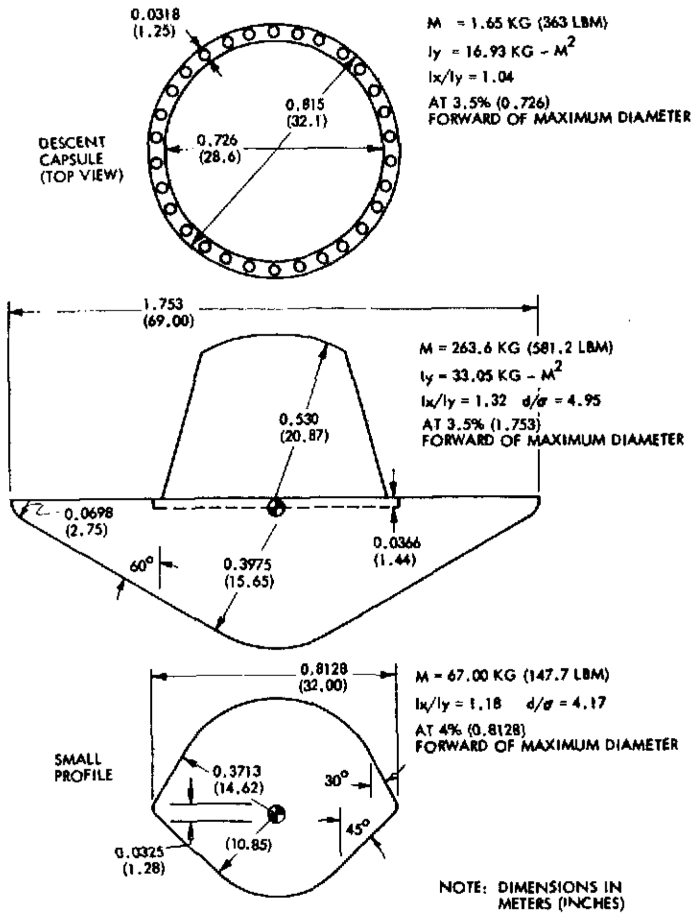


Figure 6-1. Atlas/Centaur Descent Capsule, Large and Small Probes

data directly to earth with very limited power, excursions in antenna pointing angle must be minimal.

The preferred probes have been selected to meet all of the above requirements. Aerodynamic characteristics have been estimated for the probes and used in flight dynamics studies to verify that satisfactory performance will result. Subsequent flights of the small probe configuration in the ARC Ballistic Range have verified its dynamic stability from M = 0.4 to 2.0 for angles of attack up to approximately 25 degrees. Also, dynamically scaled models of all three configurations have been flown in the NASA LRC Spin Tunnel in order to document stability characteristics under subsonic flight conditions. These tests involved perturbing the models and photographing the ensuing behavior. In this way the steady-state behavior and the degree of angular disturbance which the models could sustain without tumbling were determined. These parameters for the preferred configurations are shown in the tabulation.

Model	Steady-State Behavior	Angle Without Tumbling (deg)	Angle For Tumbling (deg)
Large Probe	Limit Cycles* $< 5^{\circ}$	Approx 50	Approx 60
Descent Capsule	Trims to zero angle of attack	Approx 90	> 90
Small Probe	Trims to zero angle of attack	45	50

*Similar behavior was observed for the small probe configuration with 10-inch diameter models but disappeared when the model diameter was increased to 23 inches. Only 10-inch diameter models of the large probe have been tested.

Six degree of freedom digital computer studies of the behavior of the large and small probes during entry have indicated that an initial angle of attack of 10 degrees will converge to less than 0.5 degree Figure 6-2. Lateral and angular g forces were shown to be less than 10 g's and 10 g/foot, respectively (Section 4).

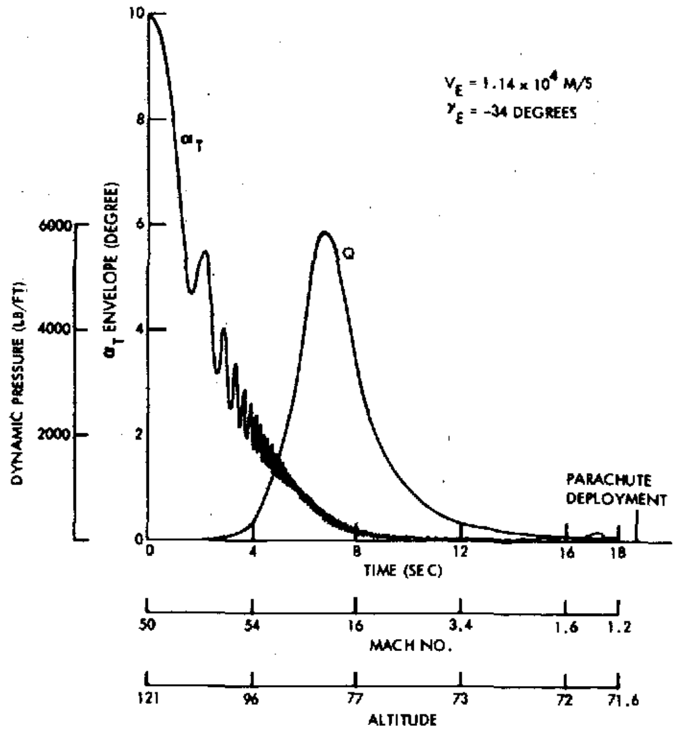


Figure 6-2. Large Probe Entry Trajectory Parameters

The effects of roll rate on the dynamic stability of the probes and the sensitivity to other aerodynamic and mass characteristics are discussed in Section 4 and 7.1. The low roll rate of 10 rpm planned for the probes is shown to be almost negligible, and variations in center of mass and principal inertia axes of 0.25 cm and 3 degrees, respectively, have minimal effect.

The responses of the probes to a horizontal wind shear profile of 0.05 mps/m up to a velocity of 100 m/s (per the NASA SP8011 Model Atmosphere) were shown in Section 4. The small probe can reach a maximum steady-state angle relative to vertical of 27 degrees and the large probe, 17 degrees. If the probe were in this condition and the wind shear suddenly terminated, i.e., the wind velocity were to become constant, these angles would represent the maximum perturbation from which the probes must be able to recover passively. These are seen to be within the probe capability indicated in the tabulation.

In the parachute descent phase, the ribless guide surface type parachute selected is very stable at zero angle of attack and has a very small lift curve slope. These features imply that the probe/parachute system will rapidly respond to wind currents with a minimum of lateral motion relative to the wind. The dynamic stability characteristics of the parachute/probe system are discussed in Section 7.5.

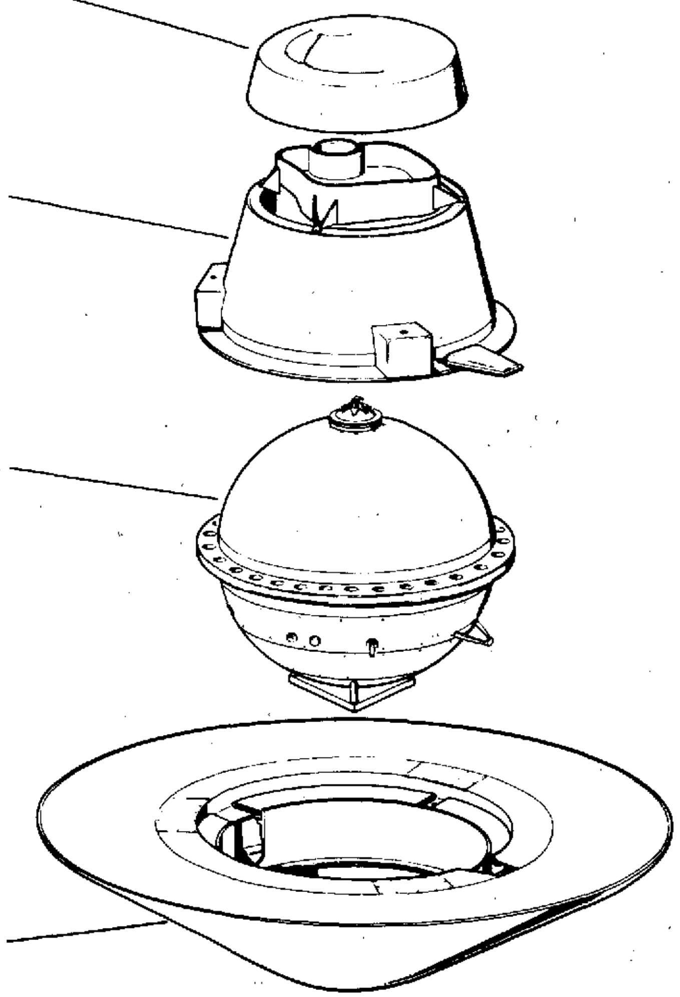
6.1.1.2 Mechanical Design Concept

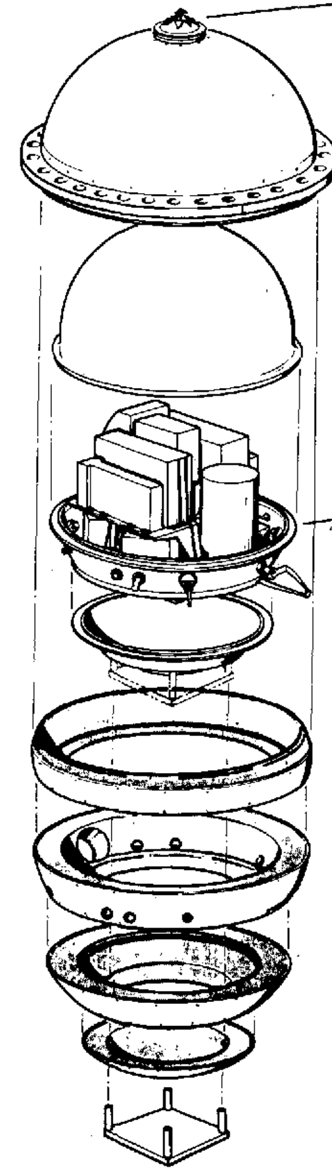
The A/C large and small probe mechanical system designs are shown in exploded views in Figures 6-3 and 6-4. Key features of these designs are summarized in this section.

Increased Safety Factors/Design Margins

Cost reductions were made in the A/C design by using weight to increase safety factors/design margins

	From	To
Structural	1.00	1.25 pressure shell
	1.25	1.56 other structure
Heat Shield	1.20	1.35 (large probe)
	1.15	1.37 (small probe)
Thermal	1.0	1.3 on insulation





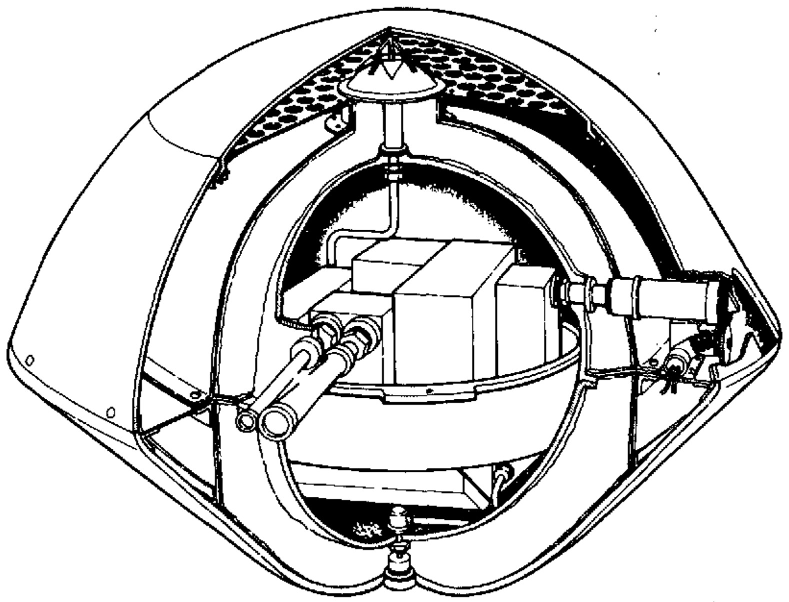
LARGE PROBE PERFORMANCE/REQUIREMENTS SUMMARY

SAFETY FACTOR	PRESSURE VESSEL	1.25/1.00
	STRUCTURAL	1.56/1.25
	HEAT SHIELD (THICKNESS)	1.35/1.20
	THERMAL INSULATION (THICKNESS)	1.3/1.0
WEIGHT	263.6 KG/ 303.2 KG	
	581.2 LB/ 668.4 LB	
THE INDICATED PERFORMANCE WEIGHT CONTAINS A 15% SCIENCE PAYLOAD CONTINGENCY		
VOLUME		
15% VOLUME CONTINGENCY IS PROVIDED FOR ALL SCIENCE INSTRUMENTS AND THOSE ELECTRONIC/ELECTRICAL ASSEMBLIES THAT ARE A NEW DESIGN		
ELECTRICAL POWER	560 W-HR/560 W-HR	
DATA	MAXIMUM RATE	128 BPS/116 BPS
	STORAGE	5120 BITS/1000 BITS
	CHANNELS	293/93
COMMAND CHANNELS	60/49	
COMMUNICATIONS	POWER AMPLIFIER	36 W/32.8W
	LINK MARGIN	5.8DB/5.4DB

Figure 6-3. Large Probe Performance Requirements

FOLDOUT FRAME 1

FOLDOUT FRAME 2



SMALL PROBE PERFORMANC/REQUIREMENTS SUMMARY

SAFETY FACTOR/ DESIGN MARGIN	PRESSURE VESSEL	1.25/1.00
	STRUCTURAL	1.56/1.25
	HEAT SHIELD (THICKNESS)	1.37/1.15
	THERMAL INSULATION (THICKNESS)	1.3/1.0
WEIGHT	70.0 KG/80.6 KG 154.4 LB/177.6 LB	
THE INDICATED PERFORMANCE WEIGHT CONTAINS A 15% SCIENCE PAYLOAD CONTINGENCY		
VOLUME		
15% VOLUME CONTINGENCY IS PROVIDED FOR ALL SCIENCE INSTRUMENTS AND THOSE ELECTRONIC/ELECTRICAL ASSEMBLIES THAT ARE A NEW DESIGN		
ELECTRICAL POWER	280 W-HR/276 W-HR	
DATA	MAXIMUM RATE	64 BPS/56 BPS
	STORAGE	5120 BITS/ 250 BITS
	CHANNELS	293/66
COMMAND CHANNELS	60/25	
COMMUNICATIONS	POWER AMPLIFIER 20 W/17.8 S	
	LINK MARGIN 5.2 DB/4.7 DB	

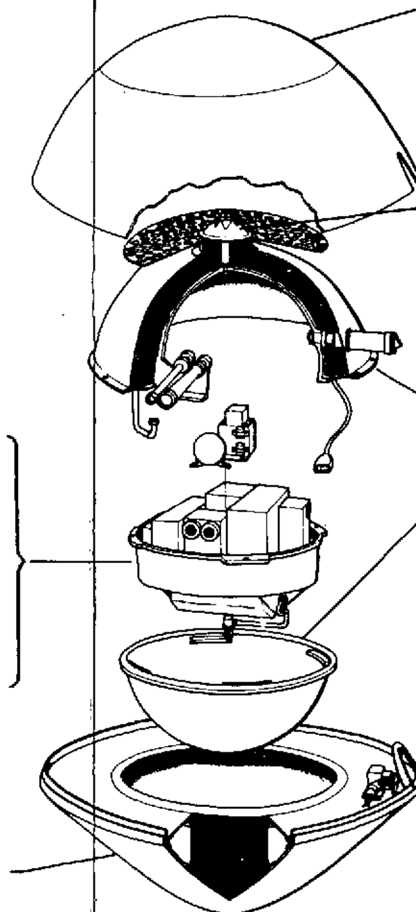


Figure 6-4. Small Probe Performance/Requirements

Increasing the structural safety factor eliminates the need for a structural test model as the qualification unit can be used for structural testing. The increased heat shield design margin allows the use of quartz nitrile phenolic as the forebody aeroshell heat shield material. This denser more predictable material reduces the entry simulation testing. The increased thermal design margin allows thicker insulation which eliminates much of the requirement for large probe thermal vacuum and descent profile simulation testing.

Packaging Approach

The large probe pressure vessel volume provides adequate space to use a modification of an existing designed Pioneer 10 and 11 digital telemetry unit, two paralleled batteries, and two paralleled power amplifiers. The small probe pressure vessel volume is sized to allow the common usage of the same digital telemetry unit, one large probe battery, and one large probe 20-watt power amplifier. The electronics components are mounted individually on an equipment shelf.

A concept using a central equipment support structure and pressure shell ring containing all the pressure shell penetrations, with an upper and lower pressure shell dome, enables maintaining the science instrument alignment requirements during manufacturing and assembly, and provides the maximum access for equipment installation, checkout, and maintanability. This concept, used in the large probe design along with a similar concept in the small probe, enables the electronics chassis and separate components to be assembled and checked out before attaching the upper and lower pressure shell domes.

Probe/Probe Bus Interface

The probe interface with the probe bus places the large probe so that its spin axis is coincident with the bus spin axis. Requirements are specified to minimize the bus c.g. offset and attendant spin balance problems. In addition, a three-point attachment utilizing ball lock fasteners (Figure 6-5) provides a symmetrical load path for the launch loads and a reliable means of attaching and releasing the probe. The probe attach points are faired in the aeroshell afterbody to minimize aerodynamic perturbations.

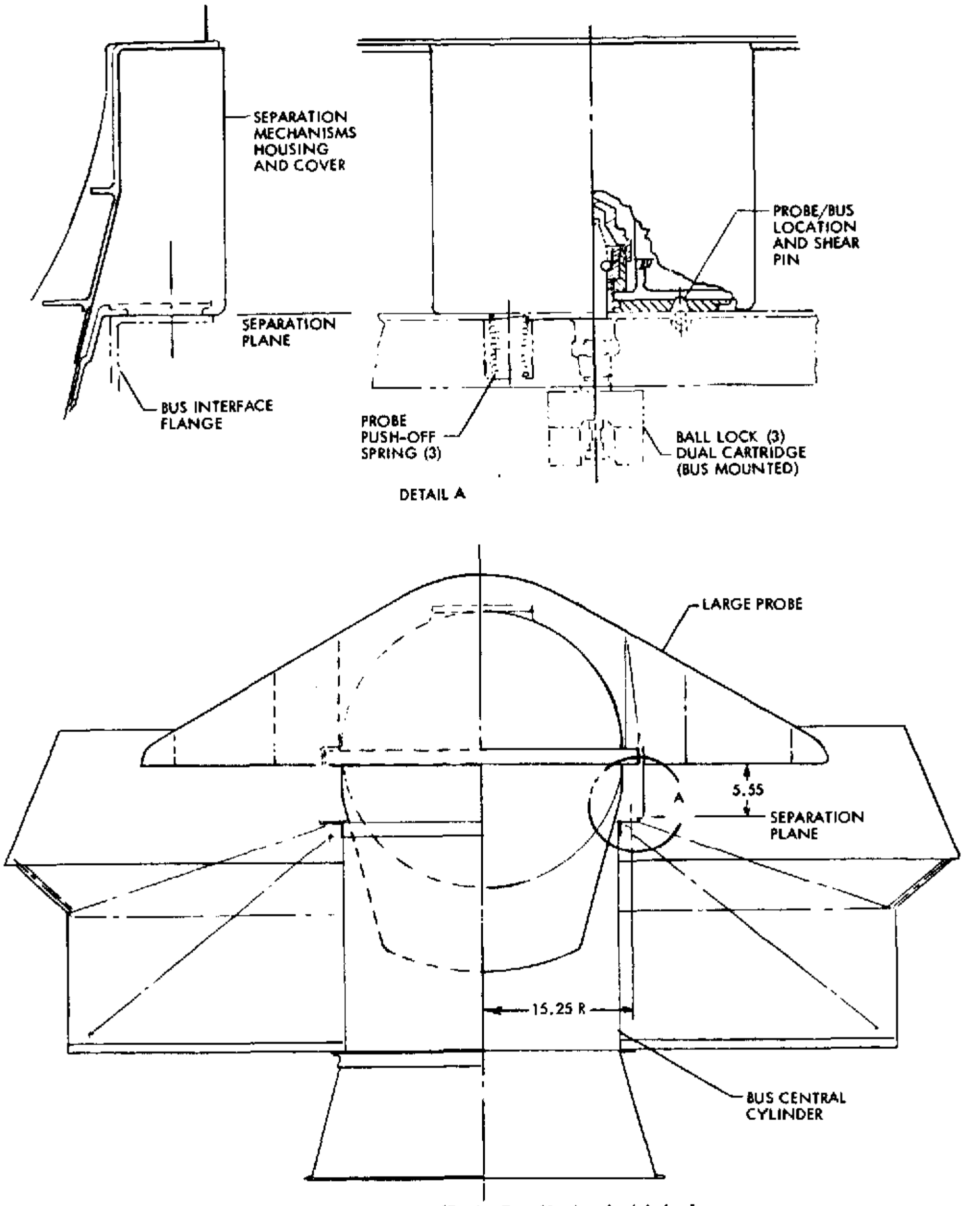


Figure 6-5. Large Probe/Probe Bus Mechanical Interface

The three small probes are mounted symmetrically about the bus spin axis. They are positioned and retained by four pairs of pads located at points approximately 1.57 radians apart around the largest diameter, with one pad from each pair bearing on the probe forebody and the other on the probe afterbody (Figure 6-6). The support pads are designed to prevent heat shield surface damage or deformation, that might significantly affect entry performance, while the small probes are restrained by the bus. The probe release system is designed to relax the preload in the support pads prior to probe release to ensure no frictional forces between the probe and bus support pads during separation. This is discussed in detail in Section 8.8.

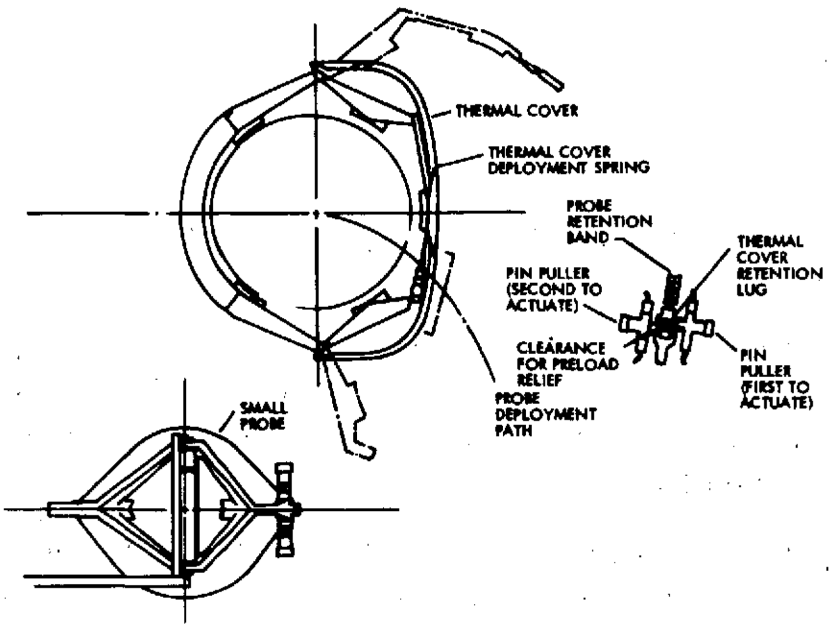


Figure 6-6. Small Probe/Probe Bus Mechanical Interface

The probe bus thermal control system maintains the probe temperature range of 255 to 339°K during the launch, cruise, and separation phases by means of control techniques using insulation, thermal coatings, thermal covers, and solar and bus heat sources.

Sequential release of the small probes permits flexibility in the acquisition of target sites to meet the science requirements while obtaining zero angle of attack for each probe. Sequential release also allows the probes to have an identical low spin rate of 1.05 rad/s (10 rpm). The probe release system (Figure 6-7) was designed to meet the attitude and velocity requirements necessary at probe release. These requirements are given in Section 5.1.

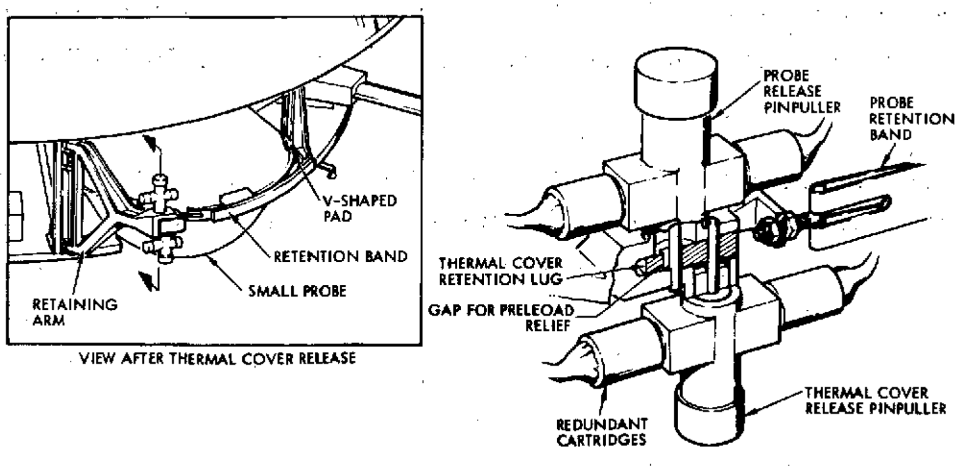


Figure 6-7. Small Probe/Probe Bus Release Mechanism

Entry and Descent Phase Design Factors

1) Entry Phase. The significant factors entering into the entry vehicle shape selection were ballistic coefficient, heating sensitivity, aerodynamic stability, and c. g. location.

The probes are designed to be aerodynamically stable and damp out yaw and pitch disturbances that are introduced by the initial angle of attack at entry. A forward location of the probe c. g. is necessary to provide adequate static stability.

Deceleration of the probes at high altitude requires a high drag, low ballistic coefficient body. Existing design blunt body shapes did not meet the probe requirements. Consequently, new designs were developed for both the large and small probes. The large probe has a 1.75 meter diameter 1.05 radian half angle forebody, and a 86.4 kg/m^2 hypersonic ballistic coefficient. The small probe has a 0.81-meter diameter, 0.78 radian half angle forebody with a cone/spherical cap afterbody and a 141.4 kg/m^2 hypersonic ballistic coefficient. These shapes were selected based on data from screening tests in the Ames Research Center water drop facility and the Langley Research Center spin tunnel facility.

Heat shield materials have been established through tests in various Venus simulation facilities. The two forebody materials selected, carbon elastomeric silicone ESA 5500 M3, and quartz nitrile phenolic, were based on flight experience in the PRIME lifting body program and AICBM hardware programs, respectively. The afterbody material is a lighter weight version of the elastomeric silicone material which has flown on PRIME and PAET vehicles.

2) Descent Phase. The major factors entering into the descent vehicle design are ballistic coefficient, science sampling requirements, aerodynamic stability, decelerator staging altitude, and the Venus environment. The large probe design uses a two-stage descent exposing the science instruments at a nominal 70 km altitude and maintaining a descent profile consistent with the science sampling requirements. The parachute ballistic coefficient is that value which is necessary to separate the descent capsule from the aeroshell while minimizing probe weight. The parachute is deployed within 30 seconds after a 50-g increasing sensor starts the descent

timer. After parachute deployment the descent velocity is reduced and decreases until parachute release approximately 40 minutes later at 42.9 km altitude. The parachute system incorporates a swivel to allow the descent capsule to spin. Large probe descent capsule aerodynamic stability is provided by a perforated ring that circles the maximum diameter.

The Venus surface environment requires an insulated pressure vessel to protect the science and electronic payloads. The externally insulated pressure shell acts as a heat sink; the combination maintains internal temperatures below $339^{\circ}K$. Thermal isolators minimize the thermal conduction from the pressure shell inward. A series of descent simulation tests have indicated that MIN-K TE 1400 is the most desirable insulation. The pressure vessel design for both the large and small probes is a machined monocoque structure with science and electronic penetrations. The pressure vessel domes incorporate a seal to maintain an internal pressure of 103 kN/m^2, which eliminates low pressure corona problems. Dry nitrogen will be used to pressurize the vessel to minimize corrosion.

The small probe design uses a single stage descent. The temperature sensor is deployed and the nephelometer and IR flux detector windows are exposed by nonexplosive pin pullers at about 70 km. The small probe has no spin control requirement. The probe is released from the bus with a 1.04 rad/s (10/rpm) spin which will vary during entry and descent.

6.1.1.3 Mass Properties

During this study mass properties estimates for the various configurations have been developed based upon the results of the subsystem studies, structural analysis, and revised layouts. These current weight estimates for the selected configuration show 263.6 kg (581.2 lb) without contingency for the large probe and 210.0 kg (463.2 lb) without contingency for three small probes, leaving a margin of 70.67 kg (155.8 lb) against the current allocation. Current weight estimates are summarized in Table 6-1. Detail component weights are presented in the subsystem description.

Complete mass properties for the various probe mission phases were calculated based upon the above evaluations and are summarized in Tables 6-2(a) and 6-2(b). All data is referenced to the coordinate systems shown in Figures 6-8 and 6-9.

Table 6-1. Atlas/Centaur Probe Current Mass (Weight) Estimate

	LARGE PROBE		SMALL PROBE	
	WT (LB)	MASS (KG)	WT (LB)	MASS (KG)
AEROSHELL	115.7	52.5	19.5	8.8
HEAT SHIELD	85.3	38.7	29.8	13.5
PRESSURE VESSEL	112.6	51.1	29.0	13.2
THERMAL CONTROL	49.5	22.5	29.8	13.5
DECELERATOR	9.7	4.4	--	--
POWER	21.7	9.8	15.2	6.9
CABLING	15.0	6.8	6.0	2.7
DATA HANDLING	8.4	3.8	8.4	3.8
RF - COMMUNICATION	11.3	5.1	4.5	2.0
ORDNANCE	12.0	5.4	1.0	0.5
AUXILIARY STRUCTURE	40.0	18.1	--	--
SCIENCE	80.0	36.3	6.6	3.0
BALANCE WEIGHTS	20.0	9.1	4.6	2.1
UNIT PROBE WEIGHT	581.2	263.6	154.4	70.0
THREE SMALL PROBES			463.2	210.0
		WT (LB)	MASS (KG)	
TOTAL PROBES		1044.4	473.6	

Table 6-2 (a). Atlas Centaur Probe Descent Mass Properties (S.I. Units)

	MASS (KG)	CENTER OF GRAVITY (CM)			MOMENT OF INERTIA ($KG-M^2$)			PRODUCT OF INERTIA (A) ($KG-M^2$)			RATIO	
		X	Y	Z	Ix	Iy	Iz	Pxy	Pxz	Pyz	Ix/Iy	Ix/Iz
LARGE PROBE												
LAUNCH THROUGH ENTRY:												
LAUNCH, CRUISE AND SEPARATION	263.6	0.254	0	0	44.314	33.473	33.931	0	0	0	1.32	1.21
POST ENTRY	254.1	-1.52	0	0	40.360	31.078	31.536	0	0	0	1.30	1.28
DESCENT:												
PARACHUTE DEPLOYMENT (B)	252.3	0.254	-0.127	0	40.304	30.670	31.077	0.141	0	0	1.31	1.30
AEROSHELL FOREBODY SEPARATION	187.1	-4.06	-0.254	0	17.843	17.151	17.556	0.126	0	0	1.04	1.02
AEROSHELL AFTERBODY SEPARATION	151.0	1.52	-0.508	0	9.510	9.418	9.772	0.288	0	0	1.01	0.97
SMALL PROBE												
LAUNCH, CRUISE AND SEPARATION	70.0	0.508	0	0	3.326	2.789	2.789	0	0	0	1.19	1.19
POST ENTRY AND DESCENT	67.0	0.254	0	0	3.052	2.579	2.579	0	0	0	1.18	1.18

NOTES: (A) INSUFFICIENT DETAIL INFORMATION TO DEVELOP PRODUCTS.

(B) PARACHUTE DEPLOYED CONDITION PROPERTIES ARE FOR THE RIGID BODY PORTION OF THE SYSTEM AS IF THE PARACHUTE WERE NOT CONNECTED.

Table 6-2 (b). Atlas/Centaur Probe Descent Mass Properties (Engineering Units)

PROBE MISSION PHASES	WEIGHT (LB)	CENTER OF GRAVITY (IN) X	Y	Z	MOMENT OF INERTIA ($LB-IN^2$) Ix	Iy	Iz	PRODUCT OF INERTIA (A) ($LB-IN^2$) Pxy	Pxz	Pyz	RATIO Ix/Iy	Ix/Iz
LARGE PROBE												
LAUNCH THROUGH ENTRY:												
LAUNCH, CRUISE AND SEPARATION	581.2	0.1	0	0	149 507	112 933	114 477	0	0	0	1.32	1.21
POST ENTRY	560.3	-0.6	0	0	136 169	104 852	106 396	0	0	0	1.30	1.28
DESCENT:												
PARACHUTE DEPLOYMENT[(B)]	556.3	0.1	-0.05	0	135 978	103 476	104 849	477	0	0	1.31	1.30
AEROSHELL FOREBODY SEPARATION	412.5	-1.6	-0.1	0	601 198	57 865	59 232	425	0	0	1.04	1.02
AEROSHELL AFTERBODY SEPARATION	333.0	0.6	-0.2	0	32 086	31 774	32 968	973	0	0	1.01	0.97
SMALL PROBE												
LAUNCH, CRUISE AND SEPARATION	154.4	0.2	0	0	11 222	9 408	9 408	0	0	0	1.19	1.19
POST ENTRY AND DESCENT	147.7	0.1	0	0	10 298	8 701	8 701	0	0	0	1.18	1.18

NOTES: (A) INSUFFICIENT DETAIL INFORMATION TO DEVELOP PRODUCTS.

(B) PARACHUTE DEPLOYED CONDITION PROPERTIES ARE FOR THE RIGID BODY PORTION OF THE SYSTEM AS IF THE PARACHUTE WERE NOT CONNECTED.

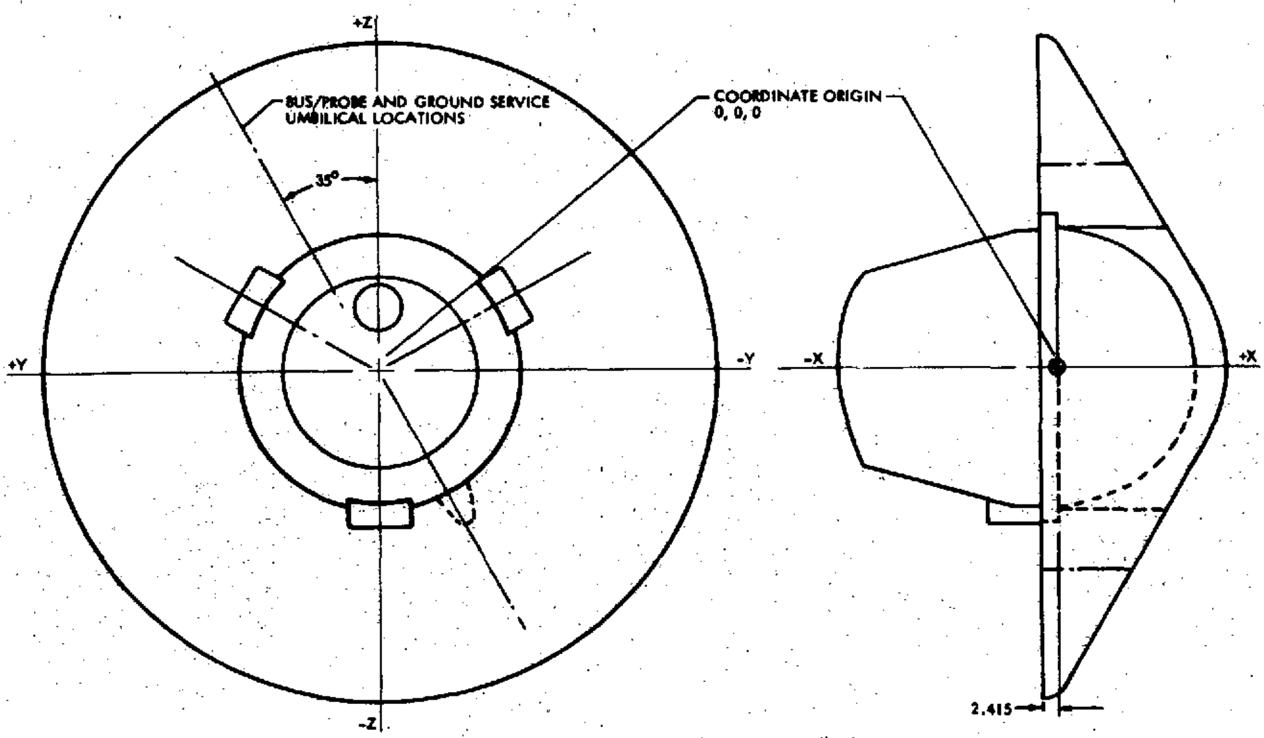


Figure 6-8. Atlas/Centaur Large Probe Reference Coordinates

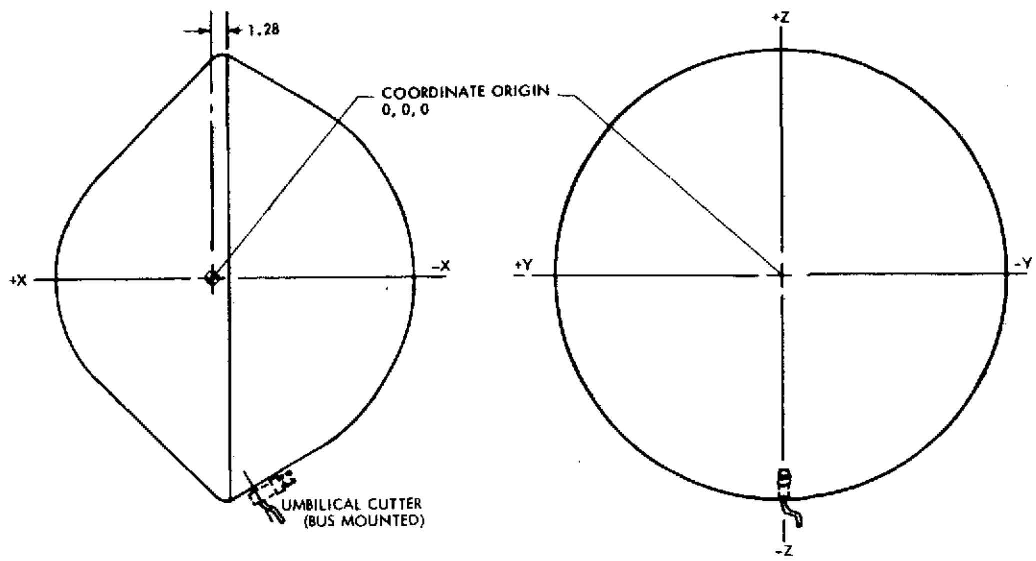


Figure 6-9. Atlas/Centaur Small Probe Reference Coordinates

Preliminary weight, center of gravity, and product of inertia limits ıd tolerances have been established. These values are defined and sum-.arized for the large and small probes in Table 6-3.

Table 6-3. Atlas/Centaur Mass Properties Limits and Tolerances for Probes

PROPERTY	LARGE PROBE	SMALL PROBE	REMARKS
MASS (WEIGHT) TOLERANCE	± 0.91 KG (± 2.0 LBM)	± 0.45 KG (± 1.0 LBM)	THE WEIGHT DIFFERENTIAL BETWEEN THE THREE SMALL PROBES SHALL NOT EXCEED 0.9 KG (2.0 LBM)
LONGITUDINAL CENTER OF GRAVITY NOMINAL LOCATION (FORWARD OF MAJOR DIAMETER CENTERLINE)	3.5%	4%	PERCENT OF MAJOR DIAMETER
DISTANCE, FORWARD MAJOR DIAMETER CENTERLINE	6.15 CM (2.42 IN.)	2.03 CM (0.8 IN.)	REQUIRED TO SATISFY AERODYNAMIC STABILITY
LONGITUDINAL CENTER OF GRAVITY TOLERANCE	± 0.51 CM (± 0.2 IN.)	± 0.254 CM (± 0.1 IN.)	REQUIRED TO SATISFY ACCELEROMETER DATA EVALUATION
RADIAL CENTER OF GRAVITY TOLERANCE TO PROBE GEOMETRIC CENTERLINE	± 0.064 CM (± 0.025 IN.)	± 0.064 CM (± 0.025 IN.)	REQUIRED TO SATISFY AERODYNAMIC STABILITY
PRODUCT OF INERTIA TOLERANCE	± 0.0038 KG-M^2 (± 13 LBM-IN.^{2})	± 0.00029 KG-M^2 (± 1.0 LBM-IN.^{2})	REQUIRED TO SATISFY SCIENCE EVALUATION
REFERENCE DATUM LOCATION	± 0.0127 CM (± 0.005 IN.)	± 0.0127 CM (± 0.005 IN.)	REQUIRED TO PROVIDE A THREE-AXIS REFERENCE FOR MEASURING THE MASS CHARACTERISTICS
PROBE GEOMETRIC CENTERLINE TOLERANCE	± 0.127 CM (± 0.050 IN.)	± 0.127 CM (± 0.050 IN.)	THE GEOMETRIC CENTERLINE OF THE PROBES WILL BE LOCATED WITH RESPECT TO THE PROBE/BUS ATTACHMENT POINTS BY USING MANUFACTURING TOOLING. THE SAME TOOLING WILL BE USED TO MACHINE THE FOREBODY HEAT SHIELD TO ACHIEVE A GEOMETRIC SHAPE SYMMETRICAL ABOUT THE DESIRED CENTERLINE.

The study has determined that ballasting will be required to obtain the mass characteristics to satisfy the probe mission. Nominal ballast weights of 2.59 kg (5.7 lb) for the large probe and 0.27 kg (0.6 lb) for each of the small probes has been assumed for this study.

6.1.1.4 Electrical Design Concept

The electrical schematic diagrams shown in Figure 6-10 (large probe) and Figure 6-11 (small probe) depict the functional relationships between probe engineering subsystems, science experiments, and the probe bus. The major electrical subsystems are the electrical power and control, data handling, and communications subsystems, outlined in heavy dashed lines.

Electrical Power

The primary +28 volt, unregulated power to operate the probes is supplied by a secondary (rechargeable) silver-zinc battery, selected because it produces the highest energy density from available space-qualified batteries.

The proposed large probe battery source consists of two batteries in parallel, each consisting of twenty series-connected sixteen ampere-hour cells. The twenty-cell configuration provides a voltage at 28 volts ±10 percent (compatible with the load requirements) for the mission life without additional regulation. Fault protection for the battery is provided by fuses in the supply lines. The final charge will be provided before launch and maintained thereafter in an open-circuit charged condition during the cruise phase. The recommended small probe battery configuration is identical to the large probe except only one battery is used.

Probe power is distributed by a power control unit and associated cabling. Isolation circuits provide compatibility with the bus power subsystem and permit operating the probes with external power for ground test and checkout. Cable cutters are used in lieu of an electrically initiated separation connector for probe bus umbilical separation. All related circuits are electrically inert before activating the cutter or are isolated to prevent damage if a short occurs. A spring-loaded connector disengages electrical connections when the large probe aeroshell afterbody is jettisoned.

Power switching functions for power transfer, ordnance, communica- ›ns, and experiments are provided by both general purpose and magnetic tching relays derived from the Viking and Pioneer programs. The relays :re selected over solid-state switches on the basis of excessive leakage rrent for solid-state switches and lower weight and cost. Satisfactory eration during sustained periods of high deceleration (typical of entry) s been demonstrated by test.

Data Handling and Command

Sensing and initiation of events are provided by a coast timer, a de- ent timer, and redundant g switches. The coast timer approach employs low-power crystal controlled oscillator design operating off the main obe battery. As a backup to the coast timer, the descent events can be itiated by the g switches which activate above 50 g's. Timing during ıtry and descent is under control of the descent timer that provides ›mmands for all discrete events. The g switches are heavily damped avoid accidental triggering of the descent sequence by the short duration ·rotechnic shocks.

The command decoder receives commands from either the bus or e descent timer and issues discrete signals for subsystem control. The ›mmands are listed in Appendix 6A.

The Data Handling and Command system controls data modes and rmats and delivers data to the bus telemetry interface or the probe com- unications subsystem for direct transmission to earth. The telemetry ıannels are listed in Appendix 6B.

The preferred design for the Atlas/Centaur configuration uses a mod- 'ied Pioneer 10 and 11 design and has fixed descent formats, a solid-state tatic memory, and convolutional coding with PCM/PSK modulation. A ummary of the basic characteristics is given in Table 6-4.

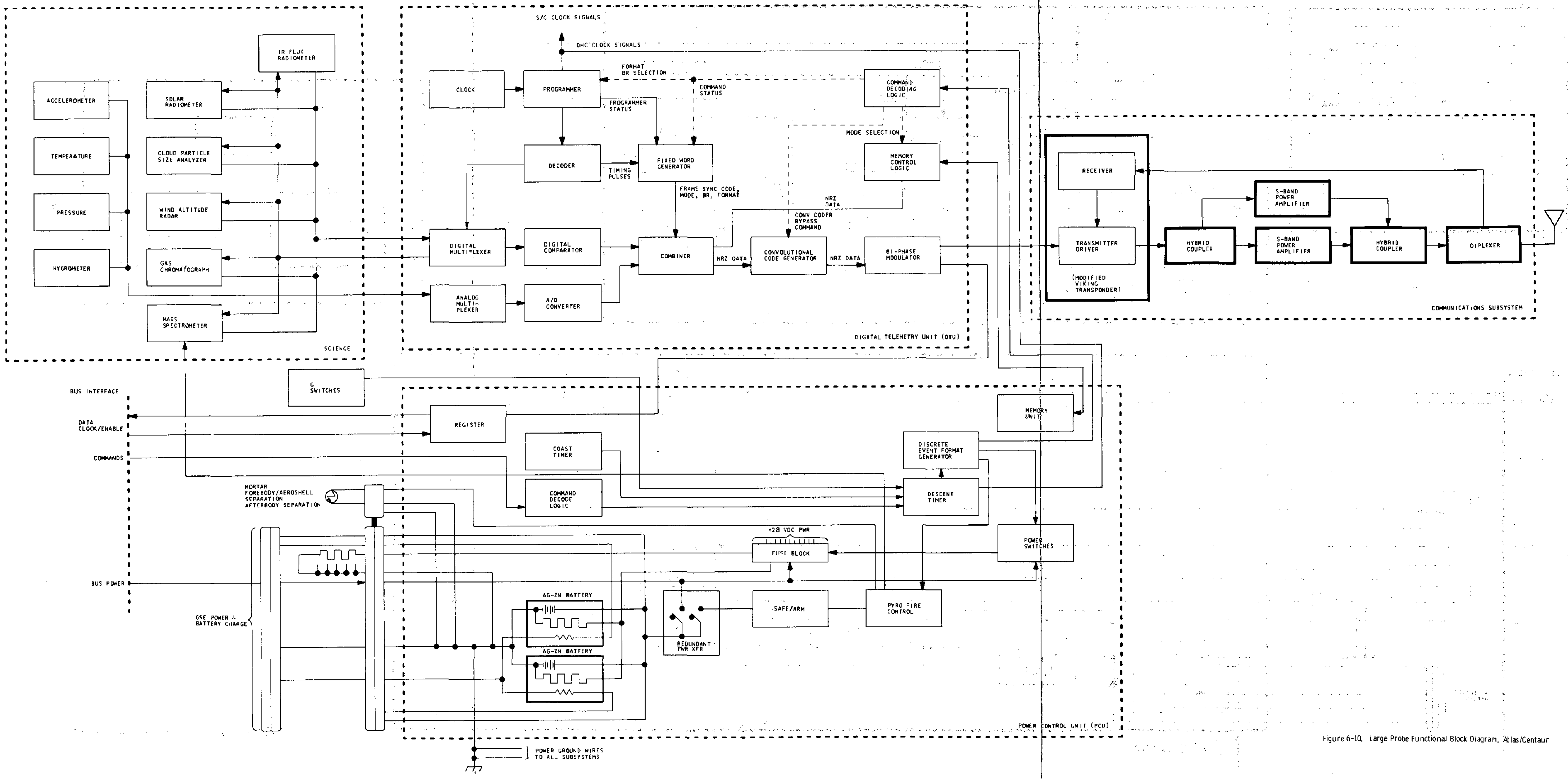


Figure 6-10. Large Probe Functional Block Diagram, Atlas/Centaur

FOLDOUT FRAME 1

FOLDOUT FRAME 2

FOLDOUT FRAME 3

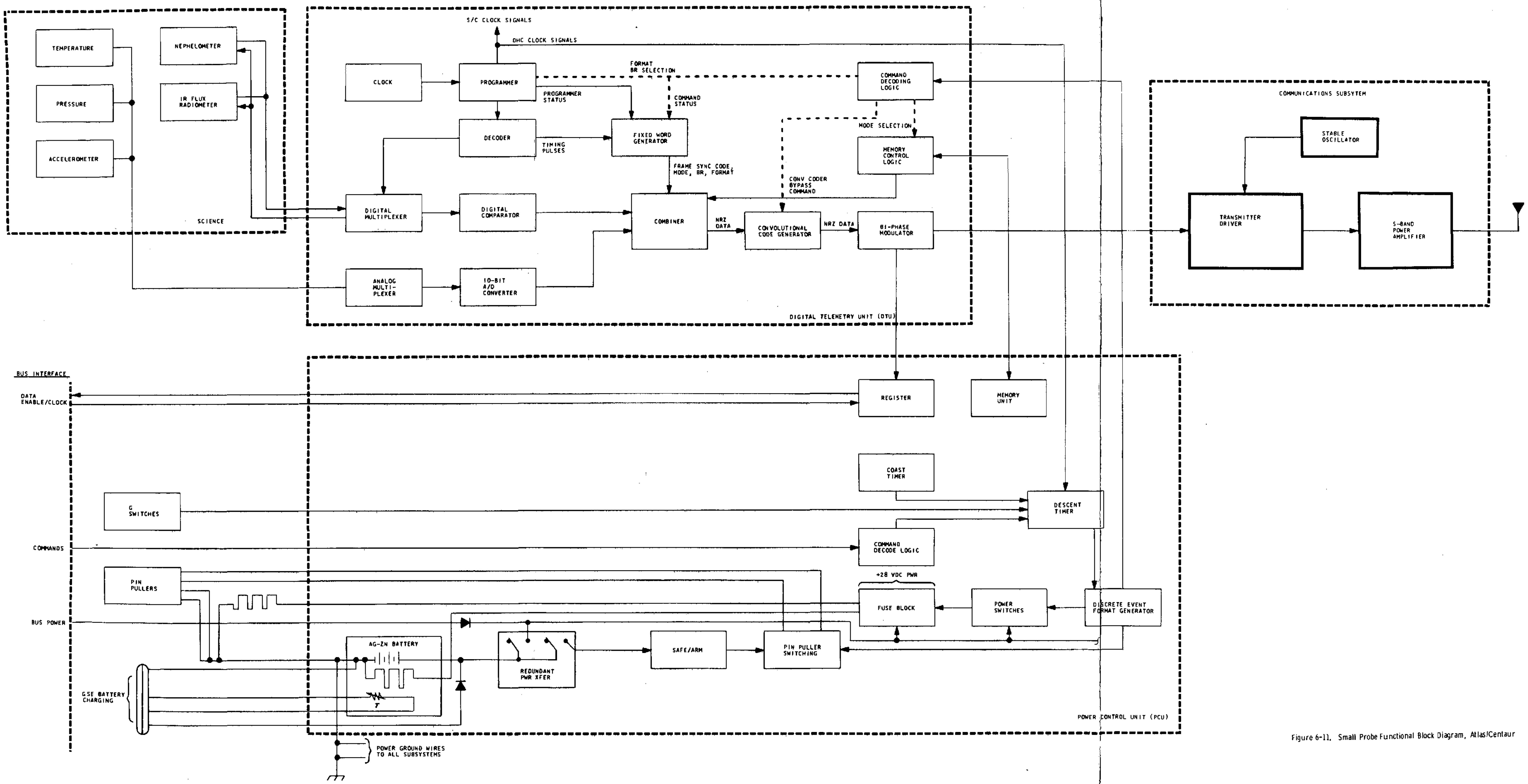


Figure 6-11. Small Probe Functional Block Diagram, Atlas/Centaur

FOLDOUT FRAME 1

FOLDOUT FRAME 2

FOLDOUT FRAME 3

Table 6-4. Data Handling Characteristics, Atlas/Centaur

	Large Probe	Small Probe
Bit Rates (bps)	128	64
Number of Formats	4	4
Number of Data Channels	246	246
Data Coding	Convolutional, Rate 1/2 (constraint length 32)	Convolutional, Rate 1/2 (constraint length 32)
Subcarrier Frequency (kHz)	32.768	32.768
Memory Size (bits)	5120	5120
Weight (kg)	3.8	3.8
Power (watts)	5.0	5.0

The memory packaged in the power control unit uses static nondestructive-readout C-MOS cells, identical to those used in the orbiter DSU. The cells are hybrid packages containing 10 chips, providing a capacity of 2560 bits per cell. The memory buffers the science data during blackout.

Data formatting is accomplished with a fixed program implemented with programmable read-only memories (PROM's). PROM's are characterized by zero cruise power, low operating power, nonvolatility, and low weight. Moreover, this approach permits format changes by replacement of PROM's without rewiring before launch (no patch board address changes are required). Clock and sync pulses are available to all science instruments at the bit rate, word rate, and frame rate, and at selected multiples of 8.

All analog-to-digital (A/D) conversion is performed in the Data Handling subsystem except from sources that are asynchronous to the basic clock or those that are spread over more than one frame. Multiplexing operations are also performed in the Data Handling subsystem.

Since all events are hardwired into the timer and sequencer, time, format, bit rate, subcommutator, identification, and memory readout are uniquely related to the frame count. No other time coding is required.

Bit rates are available in binary steps from 8 to 1024 bps. We recommend using 128 bps for all transmissions from the large probe and 64 bps for the small probes, which is compatible with the existing capability.

Communications

The recommended Communications subsystem configurations for the large and small probes utilize PCM/PSK/PM modulation because major cost savings accrue from compatibility with the existing DSN interface.

A 20-watt power amplifier module will be used for both the large and small probes with 2 units hybrid-coupled to provide the required output power (36 watts) for the large probe. Twenty watts exceeds the small probe minimum link requirements but is cost-effective since using common equipment avoids the procurement of a second design.

The antenna design is adapted from an existing Viking design. The turnstile-over-conical cavity approach offers the following features:

- Best coverage for preentry and postentry requirements
- Excellent power handling capability (75 watts)
- Good mechanical compatibility; short vertical dimension
- Low weight (0.23 kg)
- Suitable for both large and small probes
- Capable of withstanding the predicted temperatures without degradation.

For the large probe the diplexer is derived from the Pioneer 10 and 11 program. The transmitter is capable of supporting 128 bps to the Venusian surface. For the two-way link the transponder (receiver unit) includes an automatic frequency search and acquisition capability to rapidly acquire the uplink frequency following entry blackout.

A 20-watt transmitter for each of the small probes supports a 64 bps telemetry link. Provision for downlink excitation from a very stable external oscillator affords one-way Doppler tracking and DLBI experiment measurements.

Probe Electrical Equipment Definition

Table 6-5 summarizes the status of the electrical equipment for the large and small probes.

Table 6-5. Atlas/Centaur Probe Equipment Definition

COMPONENT	LP	SP	MANUFACTURER	PROGRAM	DESIGN
ANTENNA	X	X	MMC	VIKING	MODIFIED
TRANSPONDER	X		PHILCO-FORD	VIKING	MODIFIED
DIPLEXER	X		WAVECOM	PIONEER 10/11	EXISTING
HYBRIDS (90°COUPLER)	X		ANAREN MODEL 10516-3	CLASSIFIED SPACE PROGRAMS	EXISTING
POWER AMPLIFIER	X	X	MSC	NEW	EXISTING PROTOTYPE
TRANSMITTER DRIVER		X	PHILCO-FORD	VIKING	DRIVER PORTION OF TRANSPONDER
DIGITAL TELEMETRY UNIT	X	X	TRW	PIONEER 10/11	MODIFIED
BATTERIES	X	X	MMC	NEW	SAME BATTERY, LARGE PROBE AND SMALL PROBE
POWER CONTROL UNIT	X	X	MMC	NEW	SAME BATTERY, LARGE PROBE AND SMALL PROBE

6.1.1.5 Preferred Probe System Operation

The way the probes function, starting from their separation from the bus at the end of the cruise phase to impact, is described below. The hardware associated with each event or operation is also identified. A summary flight profile is given in Figure 6-12 to help correlate similar functions on each of the four probes. A detailed mission sequence of operations is contained in Appendix 6C.

Separation

After the final prelaunch checkout, probe power is off throughout the launch and most of the interplanetary cruise phase. During cruise, intermittent probe health checks can be made by an interrogation via the probe bus command and telemetry links and using probe bus electrical power. Before release, each probe will receive a preseparation status check to obtain a performance baseline. This will use both bus and probe power to exercise as much of the probe engineering subsystems and science instruments as possible. Time is allowed to perform two checkout cycles with data returned to the ground stations for analysis between these two cycles.

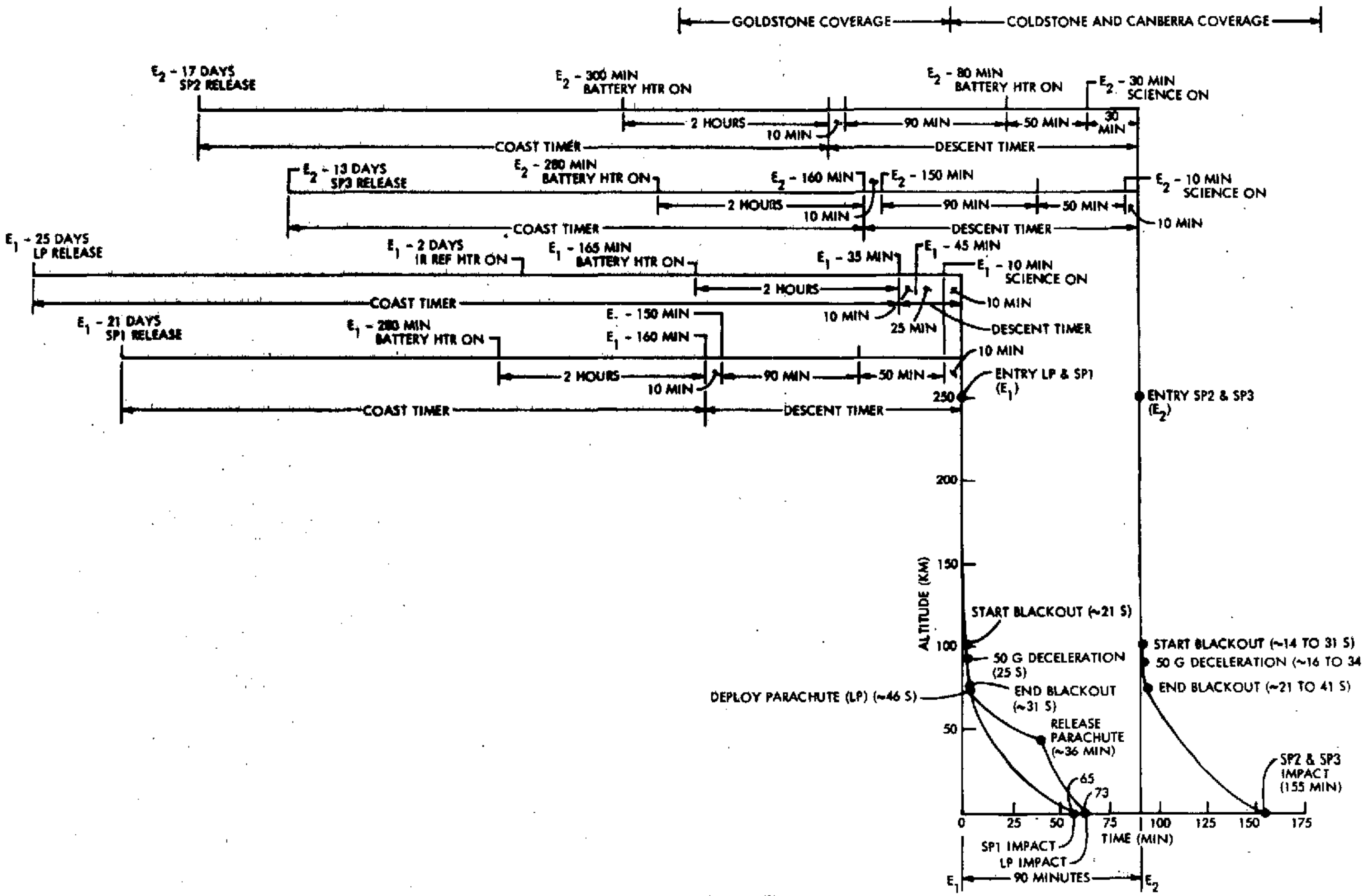


Figure 6-12. Probe Mission Profile

Just before separation, the 25-day coast timers are initiated by bus command via the umbilical connection to the probes. This timer is the only item in operation during most of the probe coast to the planet and is powered by the main probe battery.

The probes are released sequentially at four-day intervals beginning with the large probe. All probes are spin stabilized at a nominal spin rate of 10 rpm (1.05 rad/s). There is a retargeting bus maneuver between each release. The primary variables of interest during the planetary approach phase are listed in Table 6-6.

Table 6-6. Probe Mission Planetary Approach Summary

	Large Probe	Small Probe 1, 2, 3
Begin Planetary Approach	(E-25 days)	(E-21, E-17, E-13 days)
Initial Radius to Center of Planet (10^6 km)	10.7	9.0 to 5.6
Earth Aspect Angle (deg) - (at separation)	138	139, 125, 145
(at entry)	145	145, 135, 152
Solar Aspect Angle (deg) - (at separation)	40	15, 28, 36
(at entry)	72	45, 43, 55
Venus Aspect Angle (deg) - (at separation)	23	26, 14, 20
(at entry)	44	61, 34, 49

Two days before entry, the IR flux radiometer reference heater in the large probe is powered-on by the coast timer as the only event planned during this phase of the free flight.

DSN Acquisition

Several hours before entry, the probe batteries are brought up to normal operating temperature in preparation for a 10-minute DSN preentry transmission checkout operation. The probe preentry transmissions will be covered by Goldstone. Goldstone and Canberra will then receive all probe transmissions from entry until impact.

At the time the probe transmitters are turned on by each of the coast timers, the probe main power switches are turned on, and the descent timers take control. From that point on, the coast timers are no longer active.

Entry

Just prior to entry, if required, each of the small probe batteries will automatically be brought up to normal operating temperatures by thermostatically controlled electrical heaters. The large probe batteries are predicted to stay within normal operating temperature since the pre-entry transmission checkout is close enough to entry. All probe batteries are sized to furnish this energy for self-heating.

The large probe and small probe 1 enter together, nominally at 17:46:00 GMT on December 17, 1978. Ninety minutes later small probes 2 and 3 entry occurs. Entry is defined to occur at 250 km altitude. At 10 minutes before the predicted time of entry, the science instruments are turned on by the descent timer for a warmup and a preentry calibration check. High Doppler rates are predicted for about 40 minutes before and for about 2 minutes after entry.

Descent

Accelerometer data are acquired starting at the time the science instruments are turned on. Ten-second blocks of accelerometer data are continuously read into memory and recycled out by each succeeding block. At 50 g's, the redundant g switches will activate to reset the descent timer and to command retention of the last 10 seconds of accelerometer data in the memory. The information stored will include data starting before a g level of 4×10^{-4}. At a fixed time after 50 g's, the descent sequence is initiated by the descent timer. For the large probe, the first event is parachute deployment followed by jettisoning of the aeroshell forebody to expose the science instruments. Science data transmission begins at the release of the aeroshell forebody. In the case of the small probes, the first event in the descent sequence is exposure and deployment of several of the science instruments.

Science data acquisition and transmission continues until probe impact and beyond if the probes survive. The large probe parachute is released at approximately 43-km altitude; at this time the wind/altitude radar is turned on and the hygrometer is turned off. The only data transmitted after impact is from the accelerometers, used for seismic measurements. All other science and the heaters are turned off at the predicted time of impact.

6.1.2 Thor/Delta Probes

The weight limitation was the main consideration in the Thor/Delta probe system configuration selection. This weight constraint had a major impact on the mechanical and electrical system designs to satisfy the system requirements. The following paragraphs discuss the mechanical and electrical system design concepts with supporting mass properties analyses. The configuration described represents the preferred Thor/Delta concept at the time of the Midterm Reviews and features of the preferred Atlas/Centaur system described in 6.1.1 would be used for any future Thor/Delta effort.

6.1.2.1 Aerodynamic and Flight Dynamic Performance

The aerodynamic configurations of the preferred Thor/Delta probes (Figures 6-13 and 6-14) were developed to meet the drag and stability requirements of the entry and descent phasis of the mission. As described in Section 6.1.1.1 for the Atlas/Centaur probes, the design aerodynamic characteristics were evolved from both existing data and similar shapes (primarily Viking data for the entry configurations) and wind tunnel test results. The aerodynamic coefficients and stability derivatives are presented in Section 7.1.

The stability performance of the Thor/Delta probes is expected to be about the same as that of the Atlas/Centaur probes (see Section 6.1.1.1), not withstanding the differences in mass and inertia properties and somewhat different geometries. The different afterbody shape for the large probe entry configurations is expected to have little effect on convengence of entry angle of attack oscillations down through transonic speeds. In the case of the descent capsule, a vented flare was selected as the stabilizing device. This flared configuration was found in wind tunnel tests to be aerodynamically stable, but did exhibit some erratic, low-amplitude limit cycle oscillations. This behavior could degrade the large probe's scientific performance. For this reason, a simple perforated ring was selected for the Atlas/Centaur descent capsule.

The Thor/Delta small probes are nearly geometrically identical to their Atlas/Centaur counterparts; thus they are expected to behave satisfactorily in dynamic stability and response to wind shears as was the case for the Atlas/Centaur small probe.

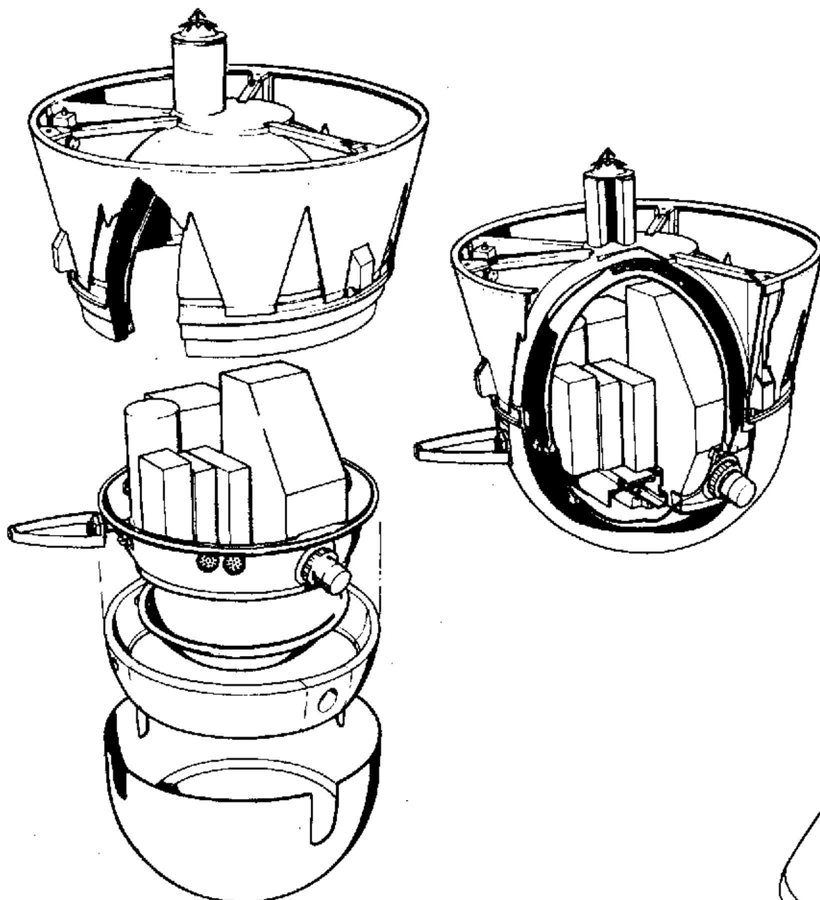


Figure 6-13. Thor/Delta Large Probe After Aeroshell Release

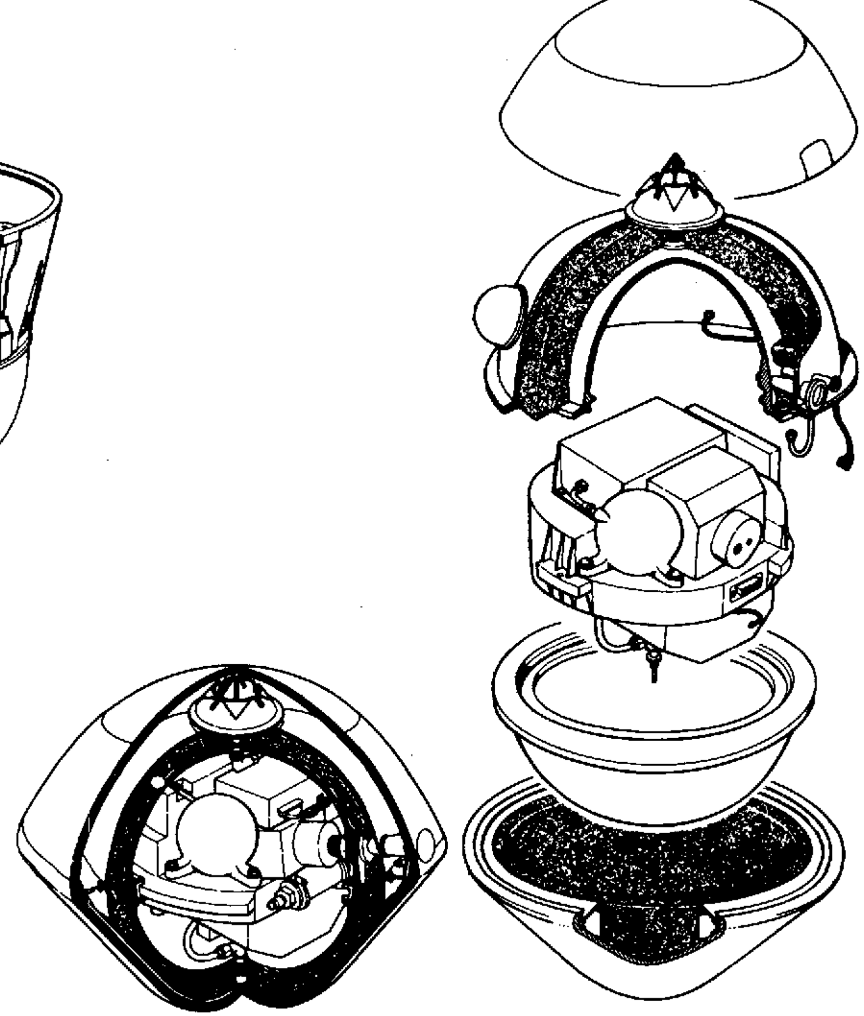


Figure 6-14. Thor/Delta Small Probe

6.1.2.2 Mechanical Design Concept

The Thor/Delta large and small probe system designs are shown in Figures 6-13 and 6-14. The weight constraints imposed on the probe design for the Thor/Delta configuration resulted in an extensive study to devise methods of packaging the large and small probes. The packaging technique is important as it affects several areas of probe design, manufacture, checkout, and maintainability. The packaging studies indicated that a concept using an integral equipment shelf and pressure shell ring, containing all the pressure shell penetrations, with an upper and lower pressure shell dome would enable manufacturing to maintain the science instrument alignment requirements and provide maximum access for equipment installation, checkout, and maintainability. This concept is used in the large probe design and a similar concept in the small probe design. Because of the pressure vessel size limitation in the small probe, a concept utilizing an integrated electronic chassis with additional mounting surfaces for separate electronic components and science instruments was adopted (Figure 6-15). With this concept it is possible to assemble and check out the electronic chassis and separate components prior to attaching the upper and lower pressure shell domes.

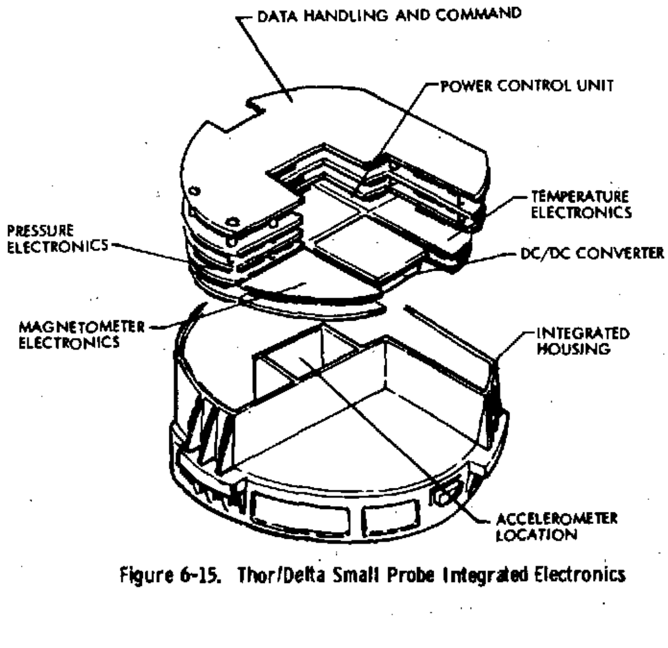


Figure 6-15. Thor/Delta Small Probe Integrated Electronics

The probe interfaces with the probe bus are essentially the same as that for the Atlas/Centaur described in Section 6.1.1.2 in the method of attachment and release and in the thermal control techniques applied to maintain temperature within proper limits. The same factors enter into entry vehicle shape selection, i.e., ballistic coefficient, heating sensitivity, aerodynamic stability, and c.g. location are the significant elements in the choice. The major factors involved in the descent vehicle design are ballistic coefficient, science sampling requirements, aerodynamic stability, decelerator staging altitude, and the Venus atmosphere.

6.1.2.3 Mass Properties

Our Phase B proposal weight estimate for the large probe was 144.5 kg (318.6 lb) and the estimate for the three small probes was 70.6 kg (155.7 lb). Because of the magnitude of uncertainty in the probe design, a major portion of the 28 kg (61.7 lb) total spacecraft contingency, or 21.5 kg (47.4 lb), was allocated to the probes, giving a total probe weight allocation of 236.6 kg (521.7 lb) (see also Section 6.2.4).

During the Phase B study, complete new mass properties estimates for the preferred Thor/Delta configurations have been developed based upon the results of the various subsystem studies, structural analysis, and revised layouts. These current weight estimates show 143.8 kg (317.3 lb) for the large probe and 76.5 kg (168.3 lb) for three small probes, leaving a margin of 15.4 kg (36.1 lb) against the original allocations. Current weight estimates are summarized in Table 6-7. Detail component weights are presented in the subsystem descriptions of this report (Section 7).

Complete mass properties for the various probe mission phases were calculated based upon these current estimates and are summarized in Table 6-8. All data is referenced to the coordinate systems shown in Figures 6-16 and 6-17.

Preliminary weight, center of gravity, and product of inertia limits and tolerances have been established. These values are defined and summarized for the large and small probes in Table 6-9.

Table 6-7. Thor/Delta Probe Current Mass (Weight) Estimate

	LARGE PROBE		SMALL PROBE	
	WT (LB)	MASS (KG)	WT (LB)	MASS (KG)
AEROSHELL	69.4	31.5	6.2	2.8
HEAT SHIELD	34.5	15.6	5.8	2.6
PRESSURE VESSEL	57.5	26.1	11.7	5.3
THERMAL CONTROL	24.5	11.1	8.0	3.6
DECELERATOR	7.1	3.2	--	--
POWER	12.2	5.5	5.7	2.6
CABLING	15.3	6.9	2.1	1.0
DATA HANDLING	5.0	2.3	4.7	2.1
RF - COMMUNICATION	4.4	2.0	3.0	1.4
ORDNANCE	2.8	1.3	--	--
AUXILIARY STRUCTURE	20.6	9.3	1.0	0.5
SCIENCE	53.0	24.0	4.9	2.2
BALANCE WEIGHTS	11.0	5.0	3.0	1.4
UNIT PROBE WEIGHT	317.3	143.8	56.1	25.5
THREE SMALL PROBES			168.3	76.5
		WT (LB)		MASS (KG)
TOTAL PROBES		584.3		220.3

The probe critical mass properties limits and tolerances are those required to satisfy aerodynamic stability. These requirements have been satisfied by establishing a method for building the probes that will provide a geometric longitudinal axis perpendicular and concentric to the separation plane within ±0.25 mm (±0.010 in.). The final machining operation on the forward heat shield will be accomplished from the same separation plane attachments. Therefore a maximum of ±0.25 mm (±0.010 in.) machining tolerances is expected from set up and cutting tolerances. Verification of the completed probe geometric center line will be accomplished by a transit or equivalent system within a ±0.25 mm (±0.010 in.) tolerance. The combination of these tolerances is expected to remain well within the ±0.050 geometric center line tolerance and at the same time satisfy the 13 lb-in.^{2} product of inertia requirement for the large probe and 1.0 lb-in.^{2} for the small probes.

The study has determined that ballasting will be required for trimming the mass characteristics so as to satisfy the probe mission mass balance requirements. Nominal ballast weights of 1.56 kg (3.5 lb) for the large probe and 0.22 kg (0.5 lb) for each of the small probe has been assumed for this study.

Table 6-8 (a). Thor/Delta Probe Descent Mass Properties (S.I. Units)

	MASS (KG)	CENTER OF GRAVITY (CM) X	Y	Z	MOMENT OF INERTIA ($KG-M^2$) Ix	Iy	Iz	PRODUCT OF INERTIA ($KG-M^2$) Pxy	Pxz	Pyz	RATIO Ix/Iy	Ix/Iz
LARGE PROBE												
LAUNCH THRU ENTRY:												
LAUNCH, CRUISE AND SEPARATION	143.9	0	0	0	15.695	12.636	12.552	-0.0008	0.004	0.156	1.24	1.25
POST ENTRY	140.8	0	0	0	14.842	12.142	12.059	-0.0008	0.004	0.156	1.22	1.23
DESCENT:												
PARACHUTE DEPLOYMENT[(A)]	139.2	0.508	0	0	14.828	11.832	11.738	0.055	0.015	0.154	1.25	1.26
AEROSHELL FOREBODY SEPARATION	98.3	-1.27	0.254	0	5.382	6.269	6.080	0.040	0.012	0.217	0.86	0.86
AEROSHELL AFTERBODY SEPARATION	88.5	2.29	0	0	4.572	4.511	4.253	0.0003	0	0.207	1.01	1.08
SMALL PROBE												
LAUNCH, CRUISE AND SEPARATION	25.4	0	0	0	0.400	0.347	0.341	0	0.0008	-0.004	1.15	1.17
POST ENTRY AND DESCENT	24.7	0	0	0	0.377	0.330	0.324	0	0.0008	-0.004	1.14	1.16

NOTE: (1) PARACHUTE DEPLOYED CONDITION PROPERTIES ARE FOR THE RIGID BODY PORTION OF THE SYSTEM AS IF THE PARACHUTE WERE NOT CONNECTED.

Table 6-8 (b). Thor/Delta Probe Descent Mass Properties (Engineering Units)

PROBE MISSION PHASES	WEIGHT (LB)	CENTER OF GRAVITY (IN) X	Y	Z	MOMENT OF INERTIA ($LB-IN^2$) Ix	Iy	Iz	PRODUCT OF INERTIA ($LB-IN^2$) Pxy	Pxz	Pyz	RATIO Ix/Iy	Ix/Iz
LARGE PROBE												
LAUNCH THRU ENTRY:												
LAUNCH, CRUISE AND SEPARATION	317.3	0	0	0	53 628	43 181	42 894	-3	14	534	1.24	1.25
POST ENTRY	310.5	0	0	0	50 716	41 490	41 203	-3	14	534	1.22	1.23
DESCENT:												
PARACHUTE DEPLOYMENT	306.8	0.2	0	0	50 669	40 432	40 111	187	46	529	1.25	1.26
AEROSHELL FOREBODY SEPARATION	216.7	-0.5	-0.1	0	18 390	21 422	20 778	134	42	742	0.86	0.86
AEROSHELL AFTERBODY SEPARATION	195.0	0.9	0	0	15 626	15 415	14 534	1	0	707	1.01	1.08
SMALL PROBE												
LAUNCH, CRUISE AND SEPARATION	56.0	0	0	0	1 366	1 187	1 167	0	3	-14	1.15	1.17
POST ENTRY AND DESCENT	54.4	0	0	0	1 291	1 130	1 109	0	3	-14	1.14	1.16

NOTE: (A) PARACHUTE DEPLOYED CONDITION PROPERTIES ARE FOR THE RIGID BODY PORTION OF THE SYSTEM AS IF THE PARACHUTE WERE NOT CONNECTED.

A series of static weight and center of gravity tests is planned to assist in minimizing the need for ballast. These test data will also be used to update the analytical model being generated for design evaluation and spin balance testing. The spin balance tests will also use the separation plane attachments, therefore providing a consistent base for defining the

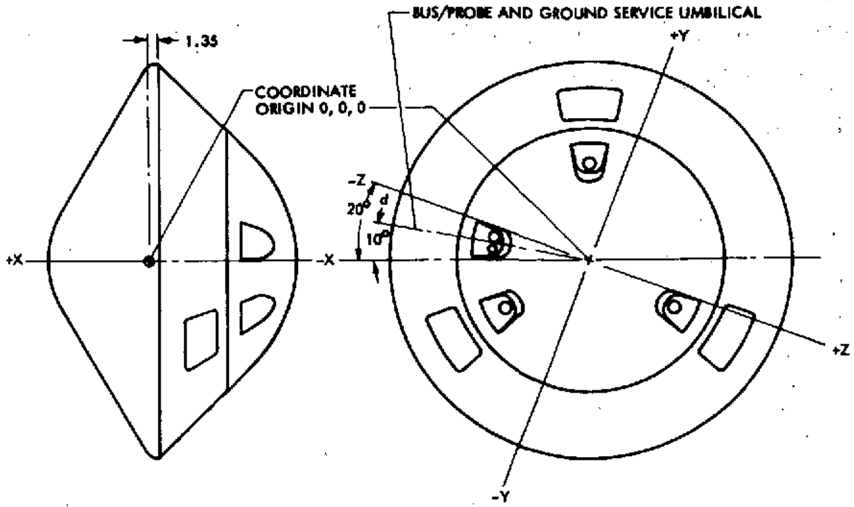


Figure 6-16. Thor/Delta Large Probe Reference Coordinates

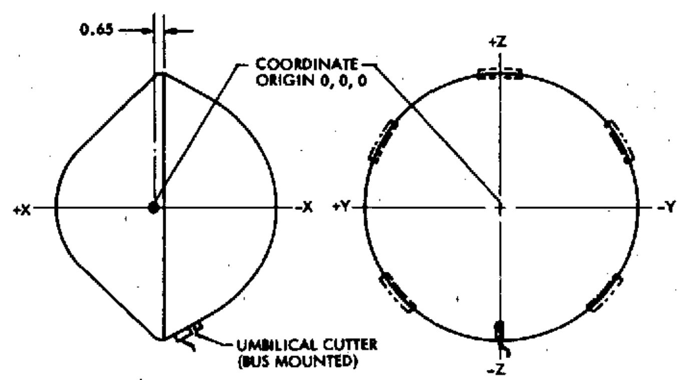


Figure 6-17. Thor/Delta Small Probe Reference Coordinates

Table 6-9. Thor/Delta Mass Properties Limits and Tolerances for Probes

PROPERTY	LARGE PROBE	SMALL PROBE	REMARKS
MASS (WEIGHT) TOLERANCE	± 0.91 KG (± 2.01 LBM)	± 0.45 KG (± 1.0 LBM)	THE WEIGHT DIFFERENTIAL BETWEEN THE THREE SMALL PROBES SHALL NOT EXCEED 0.9 KG (2.0 LBM)
LONGITUDINAL CENTER OF GRAVITY NOMINAL LOCATION (FORWARD OF MAJOR DIAMETER CENTERLINE)	2.5% (2.5%)	3.5% (3.5%)	PERCENT OF MAJOR DIAMETER
DISTANCE, FORWARD MAJOR DIAMETER CENTERLINE	3.38 CM (1.35 IN.)	1.65 CM (0.65 IN.)	REQUIRED TO SATISFY AERODYNAMIC STABILITY
LONGITUDINAL CENTER OF GRAVITY TOLERANCE	± 0.51 CM (± 0.2 IN.)	± 0.254 CM (± 0.1 IN.)	REQUIRED TO SATISFY ACCELEROMETER DATA EVALUATION
RADIAL CENTER OF GRAVITY TOLERANCE GEOMETRIES	± 0.064 CM (± 0.025 IN.)	± 0.064 CM (± 0.025 IN.)	REQUIRED TO SATISFY AERODYNAMIC STABILITY
PRODUCT OF INERTIA TOLERANCE	± 0.0038 KG-M^2 (± 13 LBM-IN.^{2})	± 0.00029 KG-M^2 (± 1.0 LBM-IN.^{2})	REQUIRED TO SATISFY SCIENCE EVALUATION
REFERENCE DATUM LOCATION	± 0.0127 CM (± 0.005 IN.)	± 0.0127 CM (± 0.005 IN.)	REQUIRED TO PROVIDE A THREE AXIS REFERENCE FOR MEASURING THE MASS CHARACTERISTICS
PROBE GEOMETRIC CENTERLINE TOLERANCE	± 0.127 CM (± 0.050 IN.)	± 0.127 CM (± 0.050 IN.)	THE GEOMETRIC CENTERLINE OF THE PROBES WILL BE LOCATED WITH RESPECT TO THE PROBE/BUS ATTACHMENT POINTS BY USING MANUFACTURING TOOLING. THE SAME TOOLING WILL BE USED TO MACHINE THE FOREBODY HEAT SHIELD TO ACHIEVE A GEOMETRIC SHAPE SYMMETRICAL ABOUT THE DESIRED CENTERLINE.

geometric centerline and the spin axis. Detail elaboration on the mass properties test plan is presented in Volume III, "Preliminary Project Development Plan."

6.1.2.4 Electrical Design Concept

The electrical schematic diagrams shown in Figure 6-18 (large probe) and Figure 6-19 (small probe) depict the functional relationships between probe engineering subsystems, science experiments, and the probe bus. The major electrical subsystems are the electrical power, data handling and command, and communications subsystems, outlined in heavy dashed lines.

Electrical Power

The primary 28-volt, unregulated power to operate the probes is supplied by a secondary (rechargeable) silver-zinc battery, selected because it produces the highest energy density from available space-qualified batteries. Moreover, the superior magnetic properties of this battery make requirements for magnetic compensation unlikely. The proposed battery consists of 20 series-connected, 9.2 ampere-hour cells. The 20-cell configuration provides a slightly higher initial voltage to maintain 28 volts ± 10 percent (compatible with the load requirements) for the mission life without additional regulation. Fault protection for the battery is provided by fuses in the supply lines. The final charge will be provided before launch and maintained thereafter in an open-circuit condition during the cruise phase.

The recommended small probe battery configuration is identical to the large probe with the substitution of 5.1 ampere-hour cells.

Probe power is distributed by a power control unit and associated cabling. Isolation circuits provide compatibility with the bus power subsystem and permit operating the probes with external power for test/checkout. Cable cutters are used, in lieu of an electrically initiated separation connector for probe bus umbilical separation and to sever electrical connections before jettisoning the large probe aeroshell forebody and afterbody.

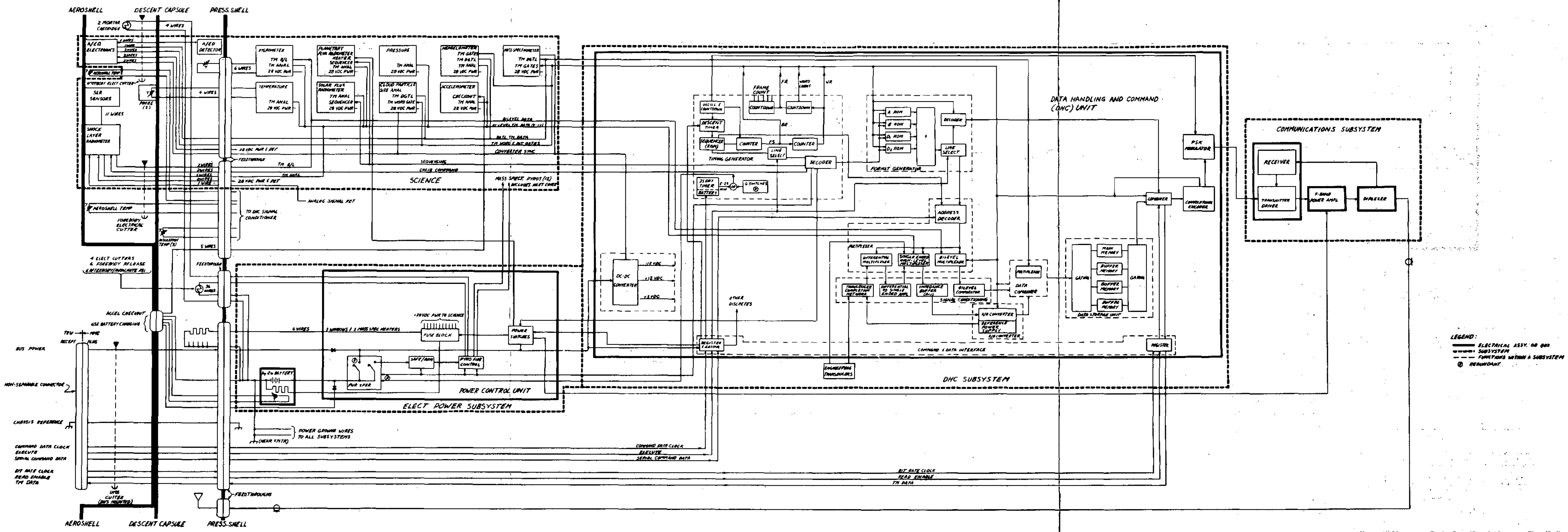


Figure 6-18. Large Probe Functional Diagram, Thor/Delta

FOLDOUT FRAME 1 FOLDOUT FRAME 2 FOLDOUT FRAME 3

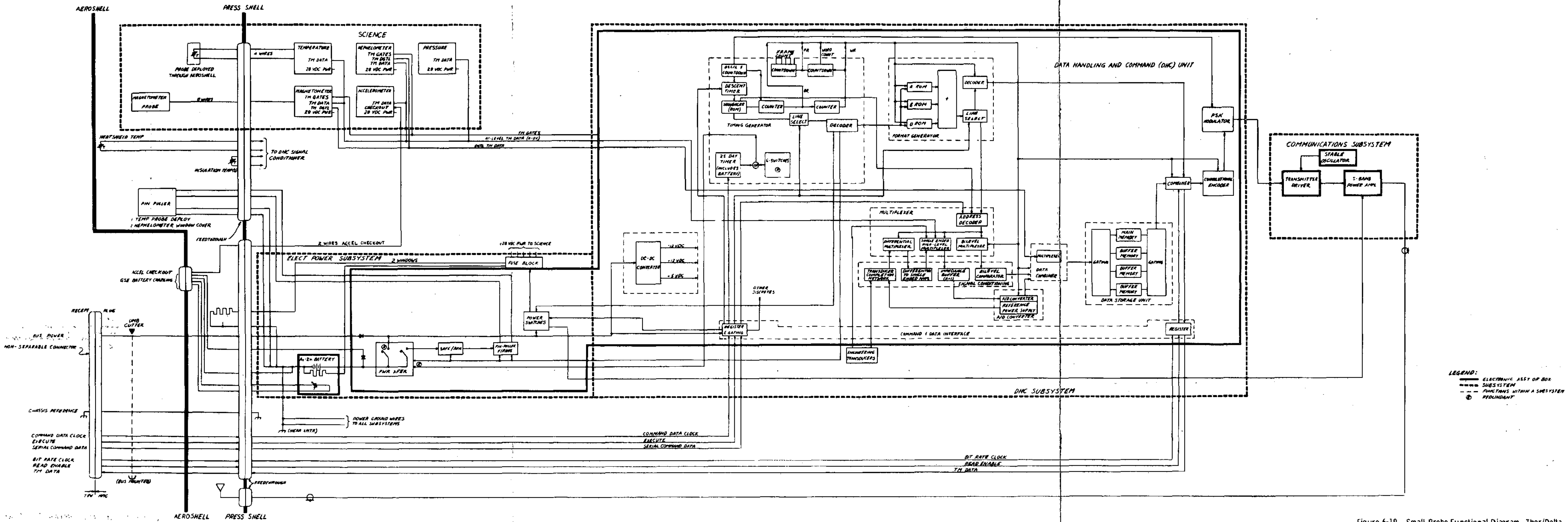


Figure 6-19. Small Probe Functional Diagram, Thor/Delta

FOLDOUT FRAME 1

FOLDOUT FRAME 2

FOLDOUT FRAME 3

All related circuits are electrically inert prior to activating the cutter, or are isolated so that no damage will result from a short circuit.

Power switching functions for power transfer, ordnance, communications, and experiments are provided by general purpose and magnetic latch relays derived from the Viking and Pioneer programs. The relays were selected over solid-state switches on the basis of excessive leakage currents and lower weight and cost. Satisfactory operation during sustained periods of high deceleration (typical of entry) have been demonstrated by test.

Data Handling and Command

The Data Handling and Command subsystem (DHC) controls data modes and formats, provides sequence control, and delivers data to the bus telemetry interface or the Communications subsystem for direct transmission to earth.

Event timing is provided by a coast timer, a descent timer, and redundant g switches. The coast timer approach employs a low-power crystal-controlled design with its own dedicated battery. As a backup the sequencing events can be initiated by the g switches which are activating above 0.5 g. Timing during entry and descent is under control of the descent timer which provides commands for all discrete events.

The g switches are disarmed until after probe separation from the probe bus to avoid accidental triggering of the descent sequence by launch or separation shocks. It should be noted that the preferred Atlas/Centaur approach of using a pneumatically damped 50-g switch eliminates the disarming requirement. The Thor/Delta arming/disarming is done by the descent timer/programmer shortly after separation.

The arming is done by a series relay switch (mag latch) driven by the descent timer (Figure 6-20). By using the descent timer, the g switch does not rely on the coast timer, the backup for g switch.

The following sequence of events is projected:

1) the probe is transferred to internal power by bus command,

2) the probe is separated,

3) the descent timer arms the g switch,

4) the descent timer turns off the main power switch.

If required, in the small probes, the descent timer can run magnetometer calibration before shutdown.

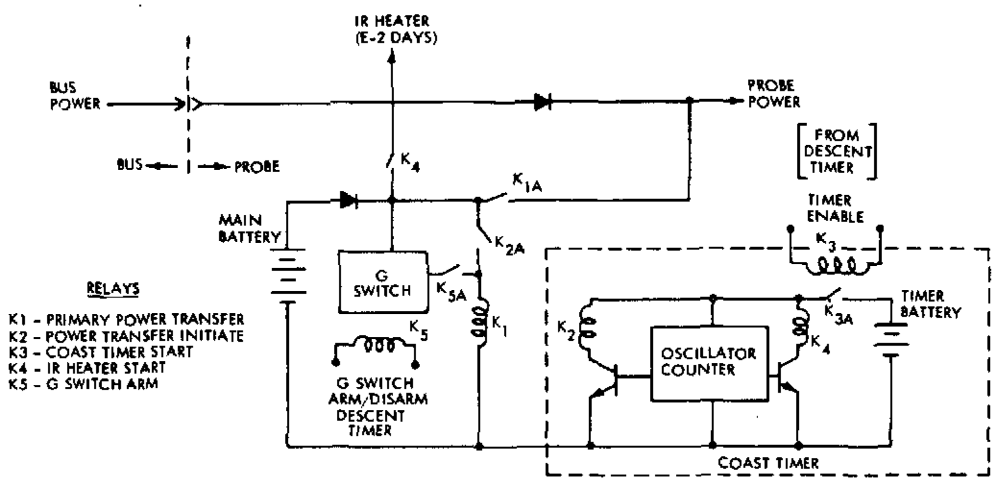


Figure 6-20. Thor/Delta Probe Power Transfer Schematic

The command decoder receives commands from either the bus or the descent timer and issues discrete signals for subsystem control. Provision is made for controlling (commanding) 60 discrete events in the large and small probes.

The preferred design for the Thor/Delta configuration uses fixed descent formats, a solid-state static memory, and convolutional coding with PCM/PSK modulation. A summary of the basic characteristics is given in Table 6-10.

The memory uses static nondestructive-readout C-MOS cells, identical to those recommended for the orbiter DSU. The cells are hybrid packages containing 10 chips, providing a capacity of 2560 bits per cell. The memory buffers the science data to match the capability of the transmission link, storing data during blackout and during times of questionable communication link integrity.

Table 6-10. Data Handling Characteristics, Thor/Delta

	Large Probe	Small Probe
Bit Rates (bps)	*1150, *128, 102.4, 85.3	*12, 10
Number of Formats	4	3
Number of Data Channels	144	96
Data Coding	Convolutional, rate 1/3 (constraint length 6)	Convolutional, rate 1/3 (constraint length 6)
Subcarrier Frequency (Hz)	16,384	16,384
Memory Size (bits)	10,240	7,680
Weight (kg)	2.3	2.1
Volume (cm^3)	2620	1980
Power (watts)	3.0	2.0

*Storage only.

Data formatting is accomplished with a fixed program implemented with programmable read-only memories (PROM's). PROM's are characterized by zero cruise power, low operate power, nonvolatility, and low weight. Moreover, this approach permits format changes by replacement of PROM's without rewiring before launch (no patch board address changes are required). Clock and sync pulses are available to all science instruments at the bit-rate, word rate, and frame rate, and at octal multiples of 512.

All analog-to-digital (A/D) conversion is performed in the DHC subsystem except from sources that are asynchronous to the basic clock or those that are spread over more than one frame. Multiplexing operations are divided between the DHC subsystem and science on the basis of minimizing interface wiring.

Viterbi decoding with a frame length of less than 1000 bits eliminates the need for a recovery sequence or "tail." Use of a Barker code sync word (7 bits) is adequate for telemetry frame synchronization, while the remaining 3 bits of a 10-bit word provide probe identification. Since all events are hardwired into the timer and sequencer, time, format, bit rate, subcom identification, and memory readout are uniquely related to the frame count; no other time coding is required.

Communications

The recommended Communications subsystem configurations for the large and small probes are shown in Figures 6-21 and 6-22, respectively. PCM/PSK/PM modulation is preferred because major cost savings accrue from the compatibility with the existing DSN interface.

The antenna design is adapted from an existing Viking design. The turnstile-over-conical cavity approach offers the following features:

- Best coverage for preentry and postentry requirements
- Excellent power handling capability (75 watts)
- Good mechanical compatibility, short vertical dimension
- Low weight (0.23 kg)
- Suitable for both large and small probes
- Capable of withstanding descent temperatures without degradation.

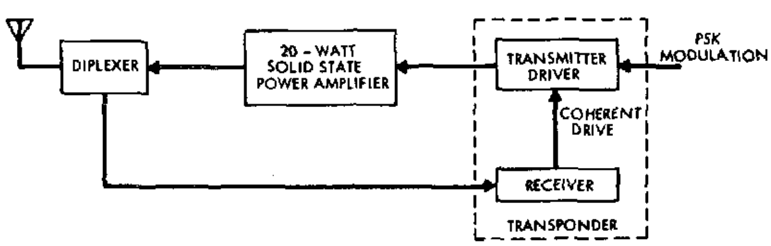


Figure 6-21. Communications Subsystem, Thor/Delta Large Probe

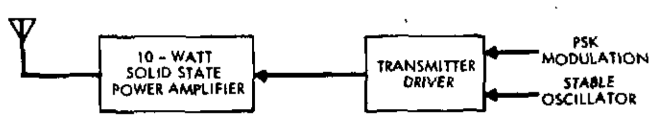


Figure 6-22. Communications Subsystem, Thor/Delta Small Probes

For the large probe the diplexer is derived from the Pioneer 10 and 11 program. The 20-watt transmitter is capable of supporting 102.4 bps at altitudes above 30 km and 85.3 bps to the Venusian surface. For the two-way link the transponder (receiver unit) includes an automatic frequency search and acquisition capability to rapidly acquire the uplink frequency before and following entry blackout.

A 10-watt transmitter for each of the small probes supports a 10-bps telemetry link. Provision for downlink excitation from a very stable external oscillator affords one-way Doppler tracking and DLBI experiment measurements.

6.1.3 MASS PROPERTIES MANAGEMENT

The management of mass properties for the probe design is unique in that the upper weight limit is defined by booster capability and the lower weight limit is defined by ballistic coefficient. The present design definition requires that the probe be within ± 1 percent of the present weight projections at the time of hardware delivery.

Successful mass properties control will be achieved by providing continuous program awareness of the weight and balance status. Management awareness of the mass properties projections, with sufficient time to react, is the key to successful mass properties management. Our approach is to --

1) Start the program with a baseline design freeze; accurately define subsystem design margins along with associated performance values; establish and publish realistic weight allocation at the subsystem level.

2) Define and establish a detail weight specification before the first design to cost review (This effort will align the weight definition with the design and build plan.); establish subsystem target weights 2 percent below the reference weights (These design and performance values will be published in the Performance Requirements Document).

3) Weight estimates derived from engineering layouts will be 100 percent calculated before the second design-to-cost review.

4) The detail weight data will be 90 to 95 percent calculated from final engineering at the completion of drawing release. To provide maximum confidence in the current mass properties data, the prototype and test hardware will be weighed and the results evaluated against previous estimates.

5) The weight data will be 80 to 90 percent actual values by the time the third design-to-cost review occurs.

6) Provide an advanced plan for managing weight margin, along with a set of cost effective solutions for under and overweight projections.

Achievement of these objectives per the milestones schedule will provide the awareness needed to cost-effectively manage the probe mass properties.

Objectives

The objectives of the mass properties control program are:

1) Minimize cost with weight,

2) Fly maximum science,

3) Deliver flight hardware that matches the allocated weight within one percent and at the same time satisfy all other associated design performance requirements.

Requirements

The mass properties requirements are described in Table 6-3. The probe weight allocations are defined by launch vehicle capability and ballistic coefficients. This resulted in the following summary weight allocations:

Weight:	Large Probe	Small Probe
Launch through entry	263.6	70.0
Postentry	254.1	67.0
Aeroshell Forebody Separation	187.1	-
Aeroshell Afterbody Separation	151.0	-

The requirements associated with center of gravity, principal axis and ballistic coefficient will require ballast. This can result in ballasting to achieve:

1) Principal axis alignment,

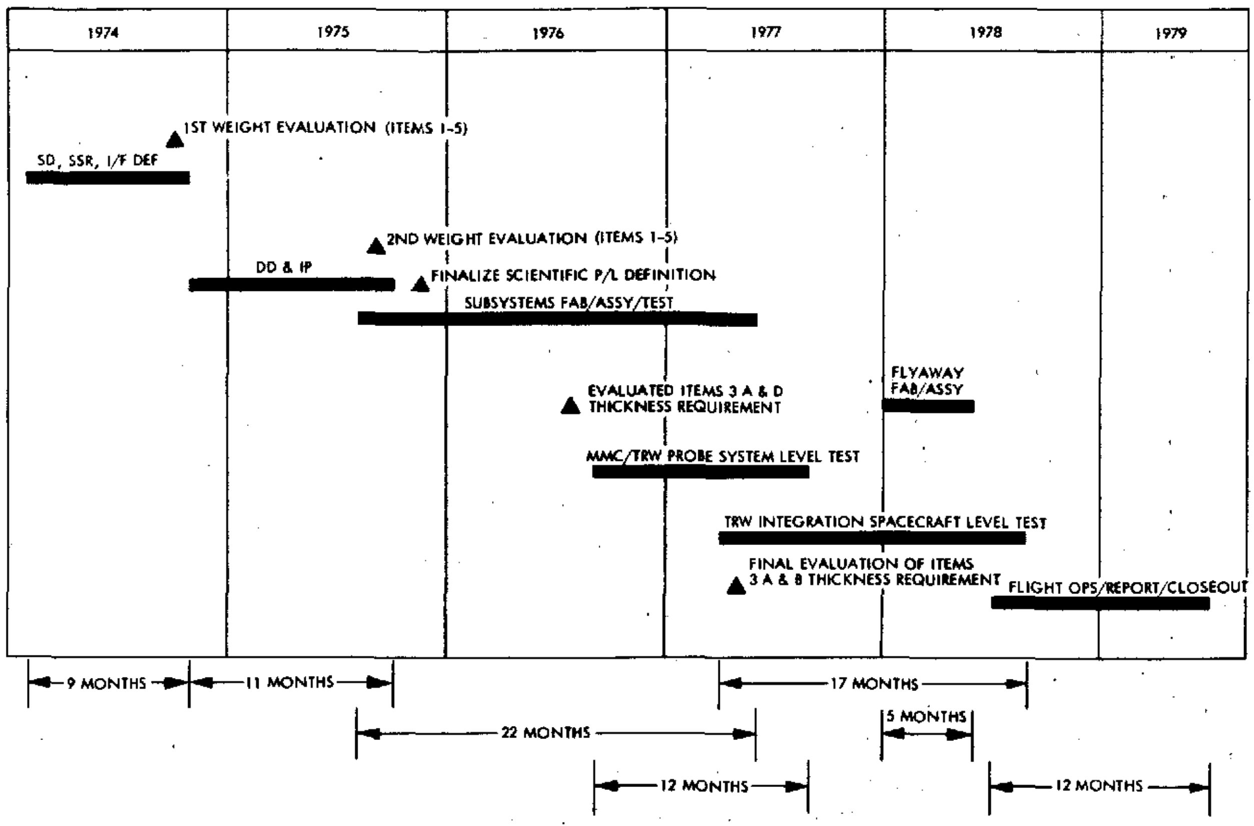
2) Radial and longitudinal center of gravity,

3) To achieve ballistic coefficient.

It is possible to ballast these three conditions in one ballast operation, but at any point in time ballast could be assigned in each category.

Weight Utilization Plan

During the program two mass properties (weight) evaluations are planned, the first being approximately October 1974 and the second being August 1975. As shown in Figure 6-23, these two evaluation periods will provide the opportunity for useful payload instead of ballast. The Atlas/ Centaur capability has allowed our design margins to be less restrictive



1) TEST ELIMINATION
   A) AEROSHELL AFTERBODY ACOUSTICS TEST: FIX - INCREASE STIFFENERS
2) MATERIAL SUBSTITUTIONS
   A) DESCENT CAPSULE OUTER SKIN: FIX - USE STEEL
   B) STABILITY RING: FIX - INSERT STEEL SLEEVES
3) MANUFACTURING TOLERANCES
   A) INSULATION THICKNESS
   B) HEAT SHIELD THICKNESS
   C) MACHINING TOLERANCE
      1) PRESSURE VESSEL
      2) LONGERONS
      3) STABILITY RING
   D) REPAIR FACTORS
4) ELECTRONICS
   A) OPEN PROCUREMENT REQUIREMENTS
5) SCIENCE
   A) ADD NEW SCIENCE
   B) REVISE EXISTING SCIENCE PARAMETERS
   C) EXCHANGE SCIENCE INSTRUMENTS

Figure 6-23. Weight Evaluation Milestone

than the Thor/Delta, therefore providing a more reliable design and the ability to minimize cost with the use of weight. The increased design margins coupled with the 15 percent contingency provides a high probability that excess weight could be made available to accommodate other candidate instruments, or further reduce cost. The growth depletion shown in Figure 6-24 provides the anticipated rate the weight margin will be used. This figure also defines the anticipated 3-σ confidence associated with the depletion projection at various program milestones. The first and second mass properties evaluations could result in adding scientific payload or it could be used to exchange a planned instrument for a more desirable instrument. This means, if alternative science is developed to the degree that a DVU could be available prior to November 1975, new or substitute alternative science could be added with minimum impact. This assumes that volume, power, environmental protection, etc. margin was allowed in the basic design on February 1974. This evaluation also clearly identifies that early in the program most changes can be implemented without causing major cost impact, but after November 1975 ballast is the probable cost-effective candidate for meeting the project weight. However, the thermal insulation and the heat shield could be varied in thickness until March 1977.

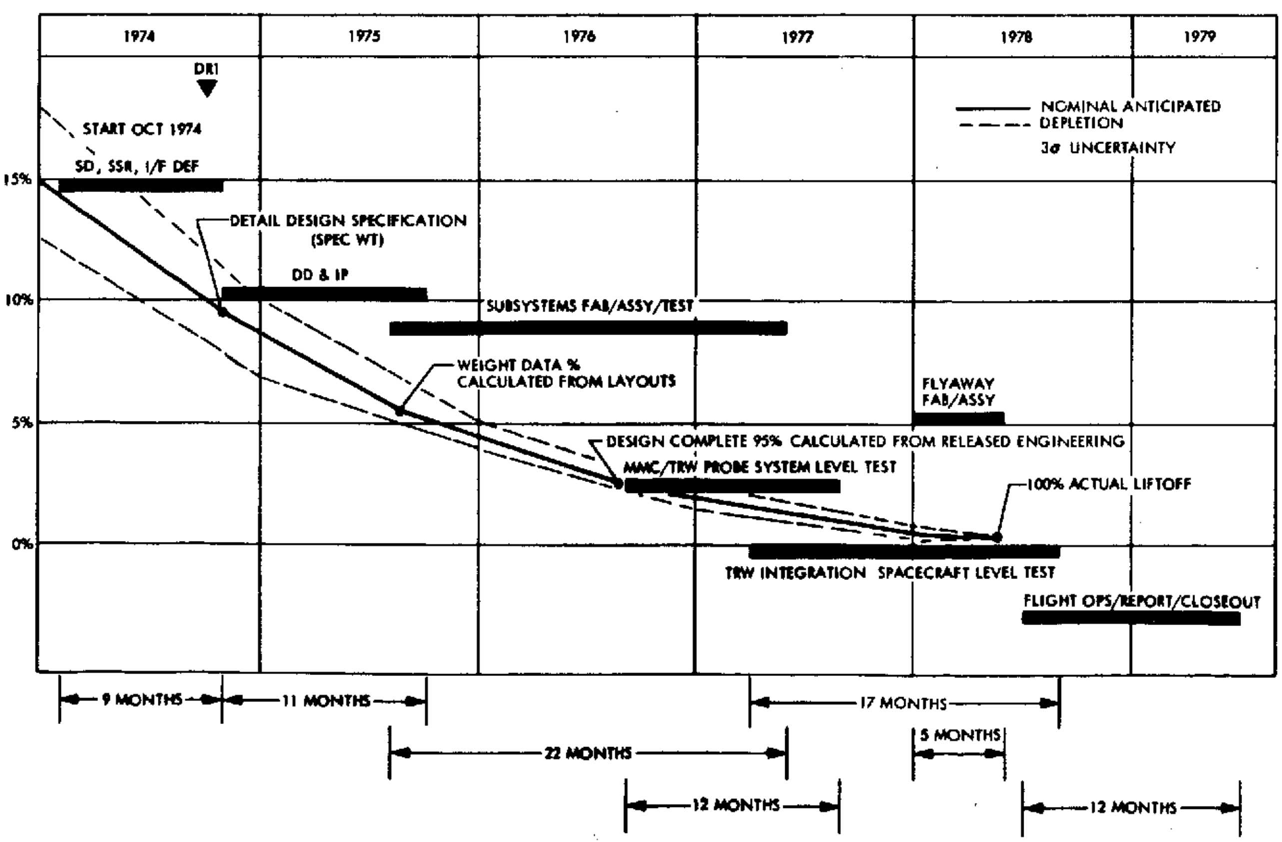


Figure 6-24. Weight Margin Depletion Schedule

The planned weight evaluations will be aligned to precede or match the design-to-cost reviews. Items currently defined as possible weight management areas are:

1) Test elimination
   a) Aeroshell afterbody acoustics test, fix - increase stiffeners
2) Material substitutions
   a) Descent capsule outer skin; fix - use steel
   b) Stability ring; fix - insert steel sleeves
3) Manufacturing tolerances
   a) Insulation thickness
   b) Heat shield thickness
   c) Machining tolerance
      1) Pressure Vessel
      2) Longerons
      3) Stability Ring
   d) Repair Factors
4) Electronics
   a) Open procurement requirements
5) Science
   a) Potential candidates changes for incorporation
      1) New science added to the payload
      2) Revisions to the science payload
      3) Science instrument exchanges
   b) Replacement science

The heat shield and insulation thickness can provide an area to whic weight can be added or removed in a cost-effective manner. These items also provide some potential for positioning the center of gravity.

The list of possible weight management areas will be evaluated and expanded as the program progresses.

6.2 Probe Bus and Orbiter System Definition

6.2 PROBE BUS AND ORBITER SYSTEM DEFINITION

6.2.1 Atlas/Centaur Configuration (Version IV Science Payload)

6.2.1.1 Mechanical Design Concept

The mechanical design concept selected for the spacecraft configuration is a conventional central cylinder/annular equipment platform configuration used on many existing spacecraft. It is a cost-effective concept, minimizing the cost and weight while meeting all program requirements.

The major loads on the spacecraft are the launch accelerations, vibrations, and acoustic environments defined in Section 5.2. The primary structure is defined by the accelerations at Centaur cutoff; for the orbiter mission, orbit insertion accelerations using the recommended solid rocket motors are as severe as the launch accelerations. The secondary structure and design of the subsystem assemblies are governed by the vibration environment which occurs during the first few seconds after liftoff. The acoustic environment at launch governs the design of any large surfaces, such as the solar array substrates.

The central cylinder is the primary load path for both the probe bus and the orbiter vehicles. The highest load is due to the large probe supported at the forward end of the central cylinder. Launch accelerations tend to buckle the central cylinder. The annular equipment platform is supported at its inner edge by the central cylinder. The small probes and the subsystem equipment on the platform are the next largest load, and the truss assembly is used to stabilize the outer periphery of the platform. This truss assembly also contains the small probe release mechanism (see Section 8.8) and supports the solar array panels. An exploded view of both probe bus and orbiter is shown in Figure 6-25.

Maintenance of structural commonality between probe bus and orbiter for a given launch vehicle was a basic ground rule of the study in order to minimize costs. Thus, a basic structure was configured, with additional features unique to each mission restriced as far as practicable to mounting provisions for mission-peculiar equipment and science instruments. This is illustrated in Figure 6-26. Notice that, except for the small probe cutouts in the equipment platform, the basic structures are identical. Even the mounting locations for the common equipment are retained.

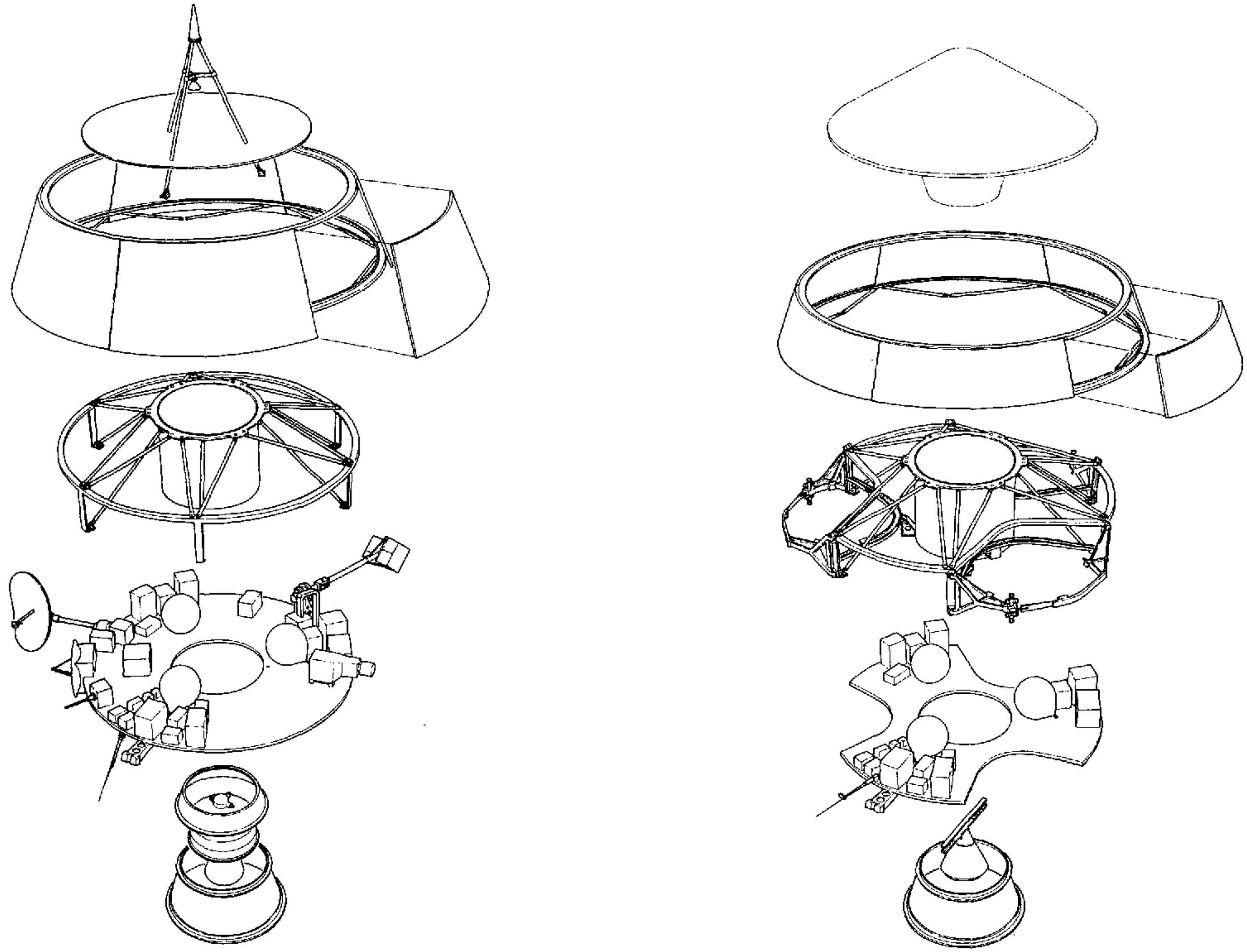


Figure 6-25. Exploded View of Orbiter and Probe Bus

A/C IV A/C IV

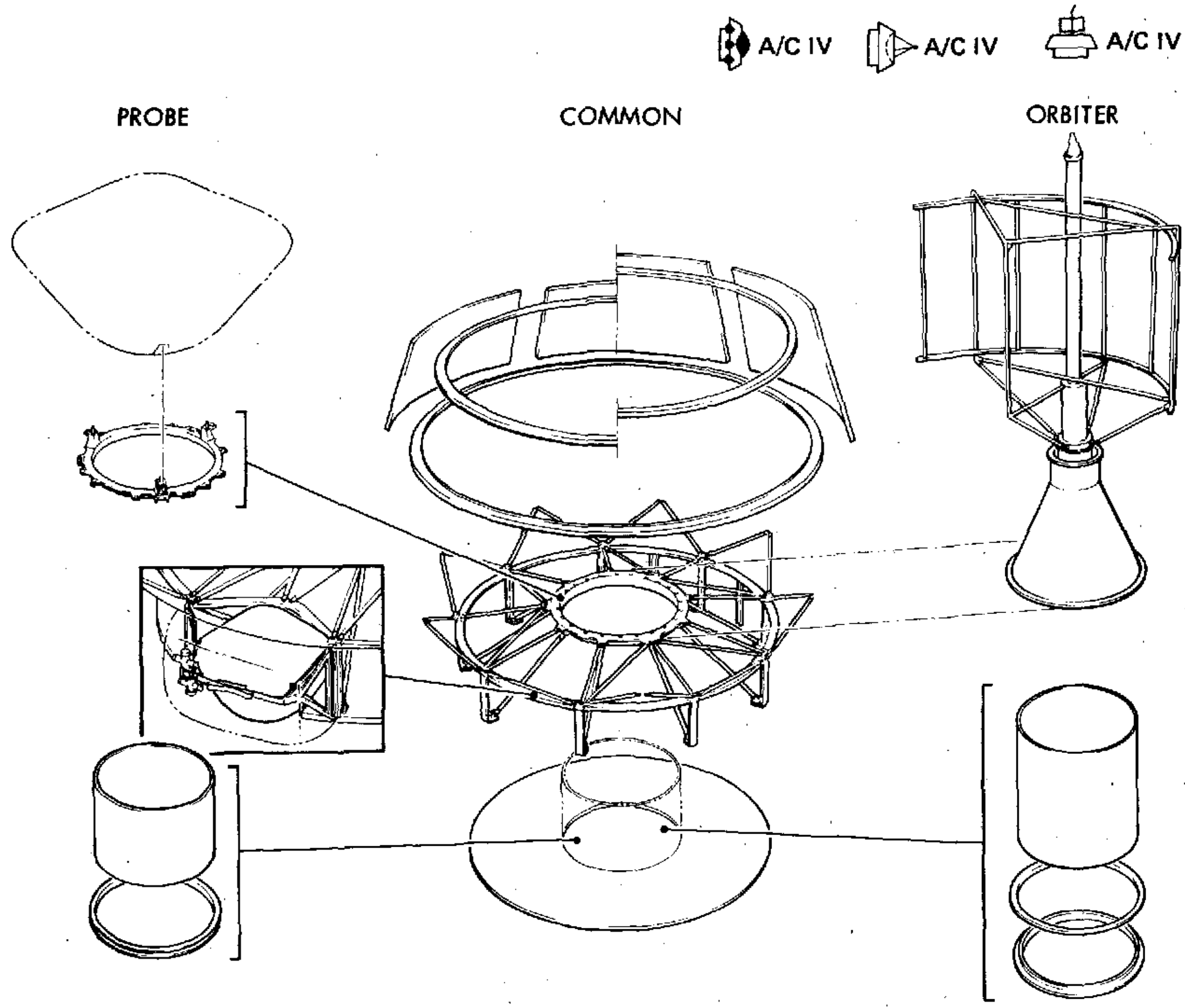


Figure 6-26. Simplicity and Commonality of Structural Design

The orbit insertion motor is mounted in the central cylinder for the orbiter mission. The resulting accelerations during orbit insertion motor firings are about the magnitude and in the same direction as the launch accelerations; thus, no new load paths are required for the orbiter spacecraft.

The science instruments are provided with their desired fields of view, as illustrated in Section 3.2. The equipment platform is mounted toward the aft end of the spacecraft, to give the spectrometer instruments clear fields of view and to avoid potential contamination of their viewing hemisphere by the attitude control system thrusters. This location of the platform yields flexibility for mounting additional science instruments by providing clear views in any aft-facing direction and by maintaining commonality with the probe bus configuration. This, is the forward portion

of each spacecraft is reserved for mission-peculiar equipment (probes for the probe bus, and communications antennas for the orbiter) and the aft portion for spacecraft equipment and science accommodation. The solar array size is also mission-related, and is forward-located.

To permit sequential small probe release it is necessary that the center of mass of the probe spacecraft after large probe release be in the same plane as the centers of mass of the small probes and spacecraft expendables. Thus, as each small probe is released, the spin axis moves parallel to the geometric axis, without tilting. This permits accurate knowledge of spacecraft attitude during small probe release which is required for good targeting knowledge, to minimize small probe wobble, and to allow the use of axial high-gain earth-pointing antennas. Therefore, the center of mass of the bus after large probe release, and of the orbiter, is just forward of the equipment platform. With the forward-located solar arrays this arrangement results in a solar torque imbalance which tends to precess the spacecraft at a steady rate. This is discussed in detail in Section 8.5. On the orbiter, in a similar fashion, a single deployable magnetometer boom is allowable by having expendables (liquid and solid propellant) and the magnetometer boom all in the same plane as the center of gravity of the spacecraft.

Each configuration has an inertia ratio well in excess of 1.10 for good spin stability during all flight phases. Mass properties of the spacecraft are tabulated in Section 6.2.1.3.

6.2.1.2 Dynamics and Attitude Control

The dynamic response of a spin-stabilized spacecraft affects the performance of the attitude control system. The primary dynamic disturbances to the spacecraft attitude are induced during the firing of the ΔV thrusters and the solid rocket motor. The thrusters are located such that velocity change (ΔV), attitude precession, and spin-rate change can be accomplished by firing appropriate pairs of thrusters (see Figure 6-27).

The axial thrusters provide ΔV by firing a pair of thrusters oriented in the same axial direction. This provides a net thrust vector which acts along the spacecraft centerline. The primary disturbance torques resulting from this are because of thrust-level differential between the two thrusters

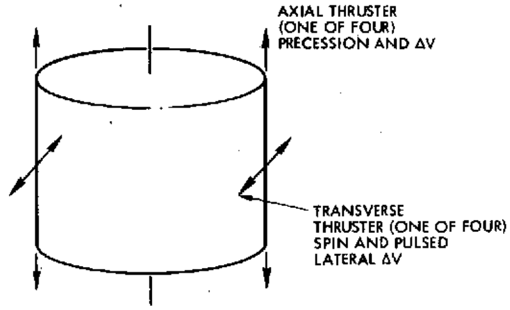


Figure 6-27. Attitude and Velocity Control Thruster Arrangement

being fired, the misalignment of the thrusters, and the spacecraft center of mass offset from the centerline. The thrust-level differential produces a body-fixed transverse torque on the spacecraft. This causes the spacecraft spin axis to cone in inertial space and produces errors in the velocity increment and spacecraft attitude.

The misalignment of the thrusters produces both a transverse torque and a spin torque on the spacecraft. The spin torque is the greater of the two effects such that for large ΔV maneuvers, the potential spin-rate change is quite large. Therefore, before a ΔV maneuver, the spin coupling should be calibrated and the maneuver then controlled such that unacceptable spin-rate changes are not induced. This can be done either by changing the initial spin rate to accommodate the expected change, or by performing the ΔV maneuver in more than one increment and correcting the spin rate after each increment.

After releasing the first and second small probes, a large center of mass offset occurs in the radial direction. This produces a transverse torque which produces the same type of disturbances as the thrust level differential produces. The resulting errors can be reduced by increasing the spin rate before the ΔV maneuvers if necessary.

The transverse (spin control) thrusters can also be used to produce a ΔV by firing in a pulse mode an appropriate pair which are both oriented in the same direction. The primary disturbance induced by this maneuver is attitude error due to axial center-of-gravity location and spin coupling due to thrust-level differential. The transverse thrusters are nominally located in the center-of-mass plane of the spacecraft. However, for the probe bus, the center-of-mass plane changes significantly after large

probe separation. Therefore, the spin control thrusters are located in the center-of-mass plane which exists after large probe separation. The attitude errors induced by a ΔV maneuver using these thrusters before large probe separation are prohibitive, thereby requiring the axial thrusters for such maneuvers.

For the orbiter configuration, the solid rocket motor firing induces dynamic errors of the same nature as those the axial thrusters produce. However, because of the much larger thrust of the solid rocket motor, the disturbance torques due to thrust vector misalignment require a much higher spin rate to limit the errors to a reasonable value. A spin rate of 6.28 rad/s (60 rpm) has been found to be necessary for this maneuver.

The magnitudes of the expected disturbances are shown in Tables 6-11 and 6-12.

Table 6-11. Atlas/Centaur Probe Mission Spacecraft Dynamic Disturbances

EVENT		ΔV (M/SEC)	THRUST EACH THRUSTER [N (LB)]	SPIN-RATE CHANGE [RAD/S (RPM)]	ANGLE-OF ATTACK ERROR [RAD (DEG)]	MOMENTUM VECTOR SHIFT [RAD (DEG)]	VELOCITY DISPERSION ANGLE [RAD (DEG)]	VELOCITY DEGRADATION (M/S)	NUTATION ANGLE [RAD (DEG)]
FIRST MIDCOURSE	(1)	13	5.2 (1.17)	±0.157 (±1.5)	0.030 (1.7)	0.007 (0.4)	0.003 (0.2)	0.0026	0.024 (1.4)
SECOND MIDCOURSE	(1)	7	5.2 (1.17)	±0.084 (±0.8)	0.030 (1.7)	0.007 (0.4)	0.003 (0.2)	0.0014	0.024 (1.4)
THIRD MIDCOURSE	(1)	2	5.2 (1.17)	±0.021 (±0.2)	0.030 (1.7)	0.007 (0.4)	0.003 (0.2)	0.0004	0.024 (1.4)
	(2)	2	5.2 (1.17)	±0.115 (±1.1)	1.920 (110)	1.920 (110)	LARGE	LARGE	
FIRST RETARGETING	(1)	1.02	5.2 (1.17)	±0.010 (±0.1)	0.017 (1)	0.007 (0.4)	0.003 (0.2)	NEGLIGIBLE	0.009 (0.5)
	(2)	1.02	5.2 (1.17)	±0.042 (±0.4)	±0.014 (±0.8)	0.014 (0.8)	NEGLIGIBLE	NEGLIGIBLE	0.003 (0.2)
SECOND RETARGETING	(1)	7.32	5.2 (1.17)	±0.063 (±0.6)	0.157 (9)	0.052 (3)	0.026 (1.5)	0.04	0.105 (6)
THIRD RETARGETING	(1)	6.34	5.2 (1.17)	±0.063 (±0.6)	0.171 (9.8)	0.082 (4.7)	0.042 (2.4)	0.04	0.108 (6.2)
	(2)	6.34	5.2 (1.17)	±0.283 (±2.7)	0.098 (5.6)	0.098 (5.6)	0.049 (2.8)	NEGLIGIBLE	0.002 (0.1)
FOURTH RETARGETING	(1)	26.55	5.2 (1.17)	±0.241 (±2.3)	0.035 (2)	0.017 (1)	0.009 (0.5)	0.01	0.017 (1)

ASSUMES 0.524 RAD/S (5 RPM) RATE FOR ALL CASES

(1) USING PAIR OF AXIAL THRUSTERS WITH 9 MILLIRADIAN MISALIGNMENT

(2) USING PAIRS OF SPIN THRUSTERS WITH ±4 PERCENT THRUST-LEVEL UNCERTAINTY

6.2.1.3 Mass Properties

Weight and mass properties estimates are summarized in this section for the Atlas/Centaur probe and orbiter spacecraft configurations. The first part presents the mass properties for the preferred Atlas/Centaur Version IV science payload and the all-1978 mission launches, and is restricted to the preferred configurations of probe and orbiter (earth-pointing) and the optional orbiter configuration with despun reflector.

Table 6-12. Atlas/Centaur Orbiter Dynamic Disturbances

EVENT	ΔV (M/SEC)	THRUST EACH THRUSTER [N (LB)]	SPIN-RATE CHANGE [RAD/S (RPM)]	ANGLE-OF ATTACK ERROR [RAD (DEG)]	MOMENTUM VECTOR SHIFT [RAD (DEG)]	VELOCITY DISPERSION ANGLE [RAD (DEG)]	VELOCITY DEGRADATION (M/SEC)	NUTATION ANGLE [RAD (DEG)]
FIRST MIDCOURSE	14.5	5.2 (1.17)	±0.293 (±2.8)	0.072 (4.1)	0.024 (1.4)	0.012 (0.7)	0.015	0.047 (2.7)
SECOND MIDCOURSE	7	5.2 (1.17)	±0.147 (±1.4)	0.072 (4.1)	0.024 (1.4)	0.012 (0.7)	0.007	0.047 (2.7)
THIRD MIDCOURSE	2	5.2 (1.17)	±0.042 (±0.4)	0.072 (4.1)	0.024 (1.4)	0.012 (0.7)	0.002	0.047 (2.7)
DEBOOST		21 000 (4800) MAX	0 (0)	0.065 (3.7)	0.021 (1.2)	0.010 (0.6)	(0.1%)	0.044 (2.5)
PERIAPSIS TRIM ΔV (TOTAL)	43.5	5.2 (1.17)	±0.785 (±7.5)	0.072 (4.1)	0.024 (1.4)	0.012 (0.7)	0.04	0.047 (2.7)

ASSUMPTIONS: 9 MILLIRADIAN THRUSTER ALIGNMENT
±4 PERCENT THRUST-LEVEL UNCERTAINTY FOR EACH THRUSTER
0.524 RAD/S (5 RPM) SPIN RATE FOR MIDCOURSE AND PERIAPSIS TRIM MANEUVERS
6.283 RAD/S (60 RPM) SPIN RATE FOR DEBOOST MANEUVER (VENUS ORBIT INSERTION)

The second part of this section is devoted to earlier studies on the Version III science payload and the 1977/1978 mission launches and includes mass properties of the preferred probe and orbiter configurations as well as three optional orbiter configurations.

The spacecraft weight summary is presented in Table 6-13 for the preferred Atlas/Centaur probe and orbiter missions. The weight margin for each mission represents the additional launch capability over the estimated current weight, including contingency and nominal science payload.

Presented below are mass properties requirements, weight summaries, detailed weight breakdowns, mass properties estimates for various flight conditions, and the coordinate reference axes and notation systems used in the mass properties analyses. Details of the large and small probe weights and mass properties used in this section are given in Section 6. 1. 1. 3.

Requirements

The requirements imposed by the Version IV science payload, the use of Atlas/Centaur for both missions, the change in the probe mission launch date to 1978, and the upward revision of anticipated Atlas/Centaur performance are reflected in Table 6-14. Based on a review of the launch vehicle supplier data, the mass properties requirements imposed by the launch vehicle are not stringent and should pose no problems.

A/C IV

A/C IV

A/C IV

Table 6-13. Spacecraft Weight Summary for the Preferred Atlas/Centaur Probe and Orbiter Missions

	ESTIMATED WEIGHT INCLUDING CONTINGENCY [KG (LB)]	
PROBE MISSION		
PROBE BUS (INCLUDING 13.7 PERCENT CONTINGENCY ON DRY SPACECRAFT)	202.8	(447)
NOMINAL SCIENCE PAYLOAD (INCLUDING 15 PERCENT CONTINGENCY)	13.6	(30)
LARGE PROBE (INCLUDING 16.5 PERCENT CONTINGENCY)	307.1	(677)
SMALL PROBES (3) (INCLUDING 16.5 PERCENT CONTINGENCY)	244.5	(539)
TOTAL SPACECRAFT AT LAUNCH	768.0	(1693)
LAUNCH CAPABILITY [EXCLUDING 47.2 KG (105 LB) PAYLOAD ADAPTER]	782.0	(1724)
WEIGHT MARGIN ABOVE CONTINGENCY ALLOWANCE	14.0	(31)
ORBITER MISSION		
SPACECRAFT (INCLUDING 13.2 PERCENT CONTINGENCY ON DRY SPACECRAFT)	405.1	(893)
NOMINAL SCIENCE PAYLOAD (INCLUDING 15 PERCENT CONTINGENCY)	45.4	(100)
TOTAL SPACECRAFT AT LAUNCH	450.5	(993)
LAUNCH CAPABILITY [EXCLUDING 47.2 KG (105 LB) PAYLOAD ADAPTER]	508.0	(1120)
WEIGHT MARGIN ABOVE CONTINGENCY ALLOWANCE	57.5	(127)

Table 6-14. Atlas/Centaur Spacecraft Mass Properties Requirements for Atlas/Centaur Version IV Science Payload and All-1978 Mission Launches

PROPERTY	REQUIREMENT	
	PROBE SPACECRAFT	ORBITER SPACECRAFT
SPACECRAFT WEIGHT AFTER SEPARATION	782 KG (1724 LB) MAXIMUM (TYPE I TRAJECTORY)	508.0 KG (1120 LB) MAXIMUM (TYPE II TRAJECTORY)
LONGITUDINAL CENTER-OF-GRAVITY LIMITATION (DISTANCE OF SPACECRAFT CENTER OF GRAVITY FORWARD OF SPACECRAFT/LAUNCH VEHICLE SEPARATION PLANE)	(NOT A STRINGENT LAUNCH VEHICLE REQUIREMENT)	
RADIAL CENTER OF GRAVITY OFFSET FROM SPACECRAFT CENTERLINE DURING LAUNCH CONDITIONS	(NOT A STRINGENT LAUNCH VEHICLE REQUIREMENT)	
INERTIA RATIO (RATIO OF SPIN TO TRANSVERSE INERTIAS FOR LONG-DURATION SPIN STABILITY CONSIDERATIONS)	>1.10	>1.10
SPACECRAFT PRINCIPAL SPIN AXIS PARALLEL TO SPACECRAFT LONGITUDINAL CENTERLINE DURING LAUNCH CONDITIONS	(NO SPECIFIC BALANCE REQUIREMENTS ARE STIPULATED BY LAUNCH VEHICLE REQUIREMENTS)	
PRINCIPAL SPIN AXIS IN THE XY AND XZ PLANE PARALLEL TO THE X-AXIS	≤0.0035 RADIAN (≤0.20 DEGREE) (PRELIMINARY ALLOCATION) DURING PROBE SEPARATION AND BUS REENTRY	≤0.0035 RADIAN (≤0.20 DEGREE) (PRELIMINARY ALLOCATION) DURING VENUS ORBIT OPERATIONS

A/C IV

A/C IV

Preferred Earth-Pointing Spacecraft Configurations

The detailed weight breakdowns for the preferred probe and orbiter spacecraft configurations are presented in Figure 6-28A. Because of the increased payload capability of the Atlas/Centaur, sufficient weight

A DETAILED WEIGHT SUMMARY

DESCRIPTION	WEIGHT: PROBE MISSION (KG)	PROBE MISSION (LB)	ORBITER MISSION (KG)	ORBITER MISSION (LB)
ELECTRICAL POWER	21.5	47.4	39.4	86.8
SOLAR ARRAY ASSEMBLY (SIX PANELS)	8.16	18.0	14.20	31.3
BATTERY	1.59	3.5	10.48	23.1
POWER CONTROL UNIT INCLUDING SHUNT	4.45	9.8	6.35	14.0
CTRF	4.99	11.0	6.03	13.3
INVERTER	2.31	5.1	2.31	5.1
COMMUNICATIONS	13.2	29.1	15.0	33.1
CONSCAN PROCESSOR	----	----	0.36	0.8
RECEIVERS (2)	4.90	10.8	2.36	5.2
TRANSMITTER DRIVERS (2)	1.27	2.8	1.09	2.4
POWER AMPLIFIERS (2)	0.54	1.2	0.54	1.2
HYBRIDS (0/2)	----	----	0.09	0.2
DIPLEXERS (2)	1.95	4.3	1.95	4.3
SWITCHES (5/7)	1.36	3.0	1.91	4.2
FORWARD OMNI	0.14	0.3	0.41	0.9
AFT OMNI	0.41	0.9	0.14	0.3
MEDIUM-GAIN ANTENNA	1.50	3.3	1.50	3.3
HIGH-GAIN ANTENNA	----	----	3.31	7.3
RF COAX AND CONNECTORS	1.13	2.5	1.36	3.0
ELECTRICAL DISTRIBUTION	15.5	34.1	15.8	34.8
COMMAND DISTRIBUTION UNIT	4.13	9.1	4.45	9.8
HARNESS AND CONNECTORS	11.34	25.0	11.34	25.0
DATA HANDLING	3.9	8.5	18.4	40.5
DIGITAL TELEMETRY UNIT	3.08	6.8	3.08	6.8
DIGITAL DECODER UNIT (2)	0.77	1.7	0.77	1.7
DATA STORAGE UNIT	----	----	14.52	32.0
ATTITUDE CONTROL	2.7	6.0	4.7	10.3
CONTROL ELECTRONICS ASSEMBLY	2.31	5.1	2.45	5.4
SUN SENSOR ASSEMBLY (2)	0.41	0.9	0.41	0.9
DRIVE SYSTEM, RAM PLATFORM	----	----	1.81	4.0
PROPULSION (DRY)	7.9	17.4	7.9	17.4
PROPELLANT TANK ASSEMBLY (3)	4.49	9.9	4.49	9.9
THRUSTER ASSEMBLY (8)	2.18	4.8	2.18	4.8
FILTER	0.23	0.5	0.23	0.5
PRESSURE TRANSDUCER	0.14	0.3	0.14	0.3
FILL AND DRAIN VALVE ASSEMBLY	0.09	0.2	0.09	0.2
PROPELLANT LINES AND MISCELLANEOUS	0.77	1.7	0.77	1.7
SOLID INSERTION MOTOR (BURNOUT)	----	----	18.7	41.1
THERMAL CONTROL	15.5	34.3	15.2	33.5
INSULATION ASSEMBLY	10.12	22.3	8.35	18.4
FORWARD CLOSURE ASSEMBLY	1.09	2.4	1.32	2.9
SIDE CLOSURE ASSEMBLY	0.54	1.2	0.54	1.2
LOUVER ASSEMBLY (3/5 SQUARE FEET)	1.77	3.9	2.95	6.5
THERMAL FIN-TRANSMITTER	0.68	1.5	0.68	1.5
HEATERS, ISOLATERS, PAINT, ETC.	1.36	3.0	1.36	3.0
STRUCTURE	75.4	166.2	73.7	162.5
CENTRAL CYLINDER ASSEMBLY	20.82	45.9	22.77	50.2
UPPER RING	(3.99)	(8.8)	(2.90)	(6.4)
CYLINDER AND LOWER FRUSTUM	(8.21)	(18.1)	(8.08)	(17.8)
PLATFORM SUPPORT RING	(1.54)	(3.4)	(1.54)	(3.4)
SEPARATION RING	(6.62)	(14.6)	(6.62)	(14.6)
MOTOR MOUNTING RING	----	----	(2.95)	(6.5)
ATTACH HARDWARE	(0.46)	(1.0)	(0.68)	(1.5)

DESCRIPTION	WEIGHT: PROBE MISSION (KG)	PROBE MISSION (LB)	ORBITER MISSION (KG)	ORBITER MISSION (LB)
STRUCTURE (CONTINUED)				
PLATFORM/COMPARTMENT ASSEMBLY	27.03	59.6	27.03	59.6
UPPER STRUTS (12)	(2.59)	(5.7)	(2.59)	(5.7)
PLATFORM STRUTS (6)	(1.68)	(3.7)	(1.68)	(3.7)
VERTICALS (6)	(1.81)	(4.0)	(1.81)	(4.0)
UPPER RING ASSEMBLY	(2.54)	(5.6)	(2.54)	(5.6)
PLATFORM STRUT FITTINGS (6)	(0.82)	(1.8)	(0.82)	(1.8)
PLATFORM ASSEMBLY	(15.69)	(34.6)	(15.69)	(34.6)
BRACKETS AND ATTACH HARDWARE	(1.90)	(4.2)	(1.90)	(4.2)
SOLAR ARRAY SUPPORT ASSEMBLY	5.90	13.0	5.81	12.8
UPPER RING	(1.64)	(3.6)	(1.55)	(3.4)
LOWER RING	(1.81)	(4.0)	(1.81)	(4.0)
STRUTS (18)	(2.45)	(5.4)	(2.45)	(5.4)
PROBE SUPPORT AND RELEASE MECHANISMS (4)	8.03	17.7	----	----
HIGH-GAIN ANTENNA SUPPORT ASSEMBLY	----	----	1.27	2.8
MAGNETOMETER BOOM ASSEMBLY	----	----	4.22	9.3
PROPULSION SUPPORT ASSEMBLY	2.27	5.0	2.27	5.0
DAMPER (2/1)	2.72	6.0	2.72	6.0
FORWARD OMNI SUPPORT	0.23	0.5	0.23	0.5
AFT OMNI SUPPORT	0.68	1.5	0.23	0.5
MEDIUM-GAIN ANTENNA SUPPORT	0.23	0.5	0.23	0.5
SCIENCE SUPPORT BRACKETRY	1.36	3.0	1.36	3.0
NM/IM SPECTROMETER SUPPORTS (2)	2.04	4.5	1.50	3.3
EQUIPMENT TIE-DOWN AND INTEGRATION HARDWARE	4.08	9.0	4.08	9.0
BALANCE WEIGHT PROVISION	5.4	12.0	5.4	12.0
• SPACECRAFT BUS LESS SCIENCE (DRY)	161.0	355.0	214.1	472.0
SCIENCE INSTRUMENTS	13.8	30.3	45.4	100.1
NEUTRAL MASS SPECTROMETER	6.26	13.8	6.26	13.8
ION MASS SPECTROMETER	1.81	4.0	1.68	3.7
ELECTRON TEMPERATURE PROBE	1.13	2.5	1.59	3.5
MAGNETOMETER	----	----	4.04	8.9
UV SPECTROMETER	3.13	6.9	6.26	13.8
SOLAR WIND ANALYZER	----	----	5.76	12.7
IR RADIOMETER	----	----	6.26	13.8
X-BAND OCCULTATION	----	----	3.13	6.9
RF ALTIMETER	----	----	10.43	23.0
RETARDING POTENTIAL ANALYZER	1.41	3.1	----	----
• SPACECRAFT BUS (DRY)	174.8	385.3	259.5	572.1
PROBES (WITHOUT CONTINGENCY)	473.7	1044.4	----	----
LARGE PROBE	263.6	581.2	----	----
SMALL PROBES (3)	210.1	463.2	----	----
• SPACECRAFT (DRY)	648.5	1429.7	259.5	572.1
PROPELLANTS AND PRESSURANT	19.9	43.8	162.6	358.4
INSERTION PROPELLANT AND EXPENDED INERTS	----	----	144.47	318.5
HYDRAZINE PROPELLANT	18.10	39.9	16.33	36.0
NITROGEN PRESSURANT	1.77	3.9	1.77	3.9
• SPACECRAFT LESS CONTINGENCY	668.4	1473.5	422.1	930.5
CONTINGENCY (NET ALLOWABLE)	113.6	250.5	85.9	189.5
(PERCENT OF DRY SPACECRAFT WEIGHT)	(17.5)		(33.2)	
• GROSS SPACECRAFT AFTER SEPARATION	782.0	1724.0	508.0	1120.0

B SUMMARY OF SPACECRAFT MASS PROPERTIES DURING VARIOUS FLIGHT CONDITIONS

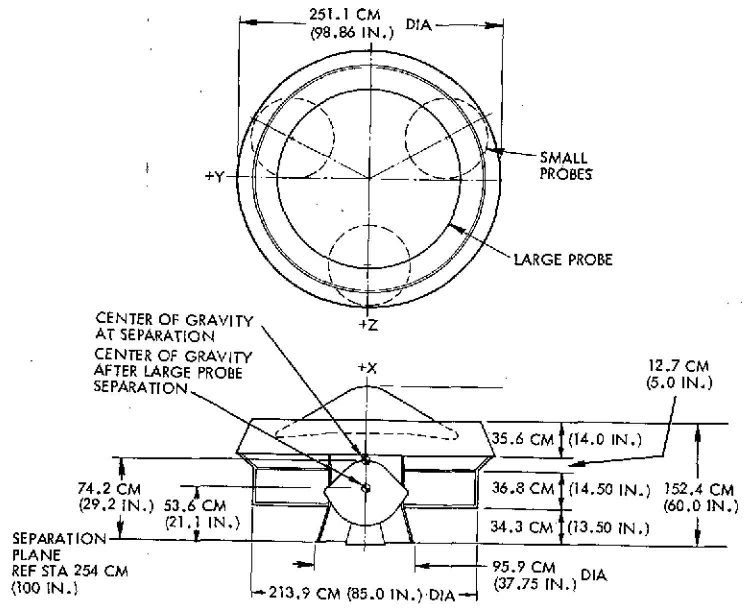
CONDITION	WEIGHT (KG)	WEIGHT (LB)	CENTER OF GRAVITY X (CM)	X (IN.)	Y (CM)	Y (IN.)	Z (CM)	Z (IN.)	MOMENTS OF INERTIA[1] I_X (SPIN) (KG-M^2)	I_X (SLUG FT^2)	I_Y (KG-M^2)	I_Y (SLUG FT^2)	I_Z (KG-M^2)	I_Z (SLUG FT^2)	PRODUCTS OF INERTIA[1] P_{XY} (KG-M^2)	P_{XY} (SLUG FT^2)	P_{XZ} (KG-M^2)	P_{XZ} (SLUG FT^2)	P_{YZ} (KG-M^2)	P_{YZ} (SLUG FT^2)	RATIO[1] I_X/I_Y	I_X/I_Z	(1)(2) λ, INERTIA PARAMETER
PROBE MISSION																							
AFTER BOOSTER SEPARATION	782.0	1724	328.2	129.2	0	0	0	0	430 (430)	317 (317)	316 (316)	233 (235)	290 (287)	214 (212)	0	0	0	0	7.7 (-15.3 DEG)	5.8	(1.35)	(1.50)	(0.42)
AFTER MIDCOURSE	772.5	1703	328.4	129.3	0	0	0	0	426 (426)	314 (314)	315 (316)	232 (233)	287 (285)	212 (210)	0	0	0	0	7.9 (-15.3 DEG)	5.8	(1.35)	(1.50)	(0.41)
AFTER LARGE PROBE SEPARATION	468.1	1032	307.6	121.1	0	0	0	0	376 (376)	277 (277)	224 (226)	165 (167)	197 (195)	145 (144)	0	0	0	0	7.9 (-15.0 DEG)	5.8	(1.66)	(1.92)	(0.78)
AFTER FIRST SMALL PROBE SEPARATION	386.5	852	307.6	121.1	15.7	6.2	9.1	3.6	297 (297)	219 (219)	202 (221)	149 (163)	137 (118)	101 (87)	0	0	0	0	-40.1 (-25.7 DEG)	-29.6	(1.34)	(2.52)	(0.73)
AFTER SECOND SMALL PROBE SEPARATION	304.4	671	307.6	121.1	0	0	23.1	9.1	226 (226)	167 (167)	169 (171)	125 (126)	103 (103)	76 (76)	0	0	0	0	7.7 (-6.6 DEG)	5.7	(1.32)	(2.20)	(0.72)
AFTER THIRD SMALL PROBE SEPARATION (END OF LIFE)	220.0	485	307.6	121.1	0	0	0	0	176 (176)	130 (130)	121 (123)	89 (91)	99 (98)	73 (72)	0	0	0	0	7.7 (-18.5 DEG)	5.7	(1.43)	(1.80)	(0.59)
ORBITER MISSION																							
AFTER BOOSTER SEPARATION (MAGNETOMETER STOWED)	508.0	1120	309.4	121.8	0	0	0	0	264 (264)	195 (195)	167 (156)	123 (115)	175 (187)	129 (138)	0	0	0	0	-15.2 (-38.1 DEG)	-11.2	(1.70)	(1.41)	(0.53)
AFTER BOOSTER SEPARATION (MAGNETOMETER DEPLOYED)	508.0	1120	309.4	121.8	1.5	0.6	-0.3	-0.1	312 (312)	230 (230)	168 (161)	124 (119)	221 (226)	163 (167)	0	0	0	0	-18.4 (-17.3 DEG)	-13.6	(1.93)	(1.38)	(0.59)
AFTER MIDCOURSE (MAGNETOMETER DEPLOYED)	505.8	1115	309.4	121.8	1.5	0.6	-0.3	-0.1	310 (310)	229 (229)	167 (161)	123 (119)	221 (226)	163 (167)	0	0	0	0	-18.4 (-17.3 DEG)	-13.6	(1.92)	(1.37)	(0.59)
PRIOR VENUS ORBIT INSERTION (MAGNETOMETER STOWED)	505.8	1115	309.4	121.8	0	0	0	0	263 (263)	194 (194)	167 (155)	123 (114)	175 (186)	129 (137)	0	0	0	0	-15.2 (-38.1 DEG)	-11.2	(1.70)	(1.42)	(0.54)
AFTER VENUS ORBIT INSERTION (MAGNETOMETER STOWED)	358.8	791	309.4	121.8	0	0	0	0	256 (256)	189 (189)	159 (148)	117 (109)	167 (179)	123 (132)	0	0	0	0	-15.2 (-38.1 DEG)	-11.2	(1.73)	(1.43)	(0.56)
AFTER VENUS ORBIT INSERTION (MAGNETOMETER DEPLOYED)	358.8	791	309.4	121.8	2.3	0.9	-0.3	-0.1	302 (302)	223 (223)	160 (153)	118 (113)	213 (218)	157 (161)	0	0	0	0	-18.4 (-17.3 DEG)	-13.6	(1.97)	(1.38)	(0.61)
AFTER SCIENCE DEPLOYMENT (MAGNETOMETER DEPLOYED)	358.8	791	309.4	121.8	4.8	1.9	2.0	0.8	325 (325)	240 (240)	182 (182)	134 (134)	213 (214)	157 (158)	0	0	0	0	-4.5 (-7.8 DEG)	-3.3	(1.79)	(1.52)	(0.64)
END OF LIFE (MAGNETOMETER DEPLOYED)	347.4	766	309.4	121.8	5.1	2.0	2.0	0.8	321 (321)	237 (237)	180 (179)	133 (132)	212 (212)	156 (156)	0	0	0	0	-4.5 (-7.9 DEG)	-3.3	(1.79	(1.52)	(0.64)

NOTE: (1) VALUES IN PARENTHESES ARE REFERENCED RELATIVE TO THE PRINCIPAL AXES

(2) INERTIA PARAMETER, $\lambda = \left[\left(\frac{I_X - I_Y}{I_Y}\right)\left(\frac{I_X - I_Z}{I_Z}\right)\right]^{0.5}$

C COORDINATE REFERENCE AXES AND NOTATION

PROBE SPACECRAFT



ORBITER SPACECRAFT

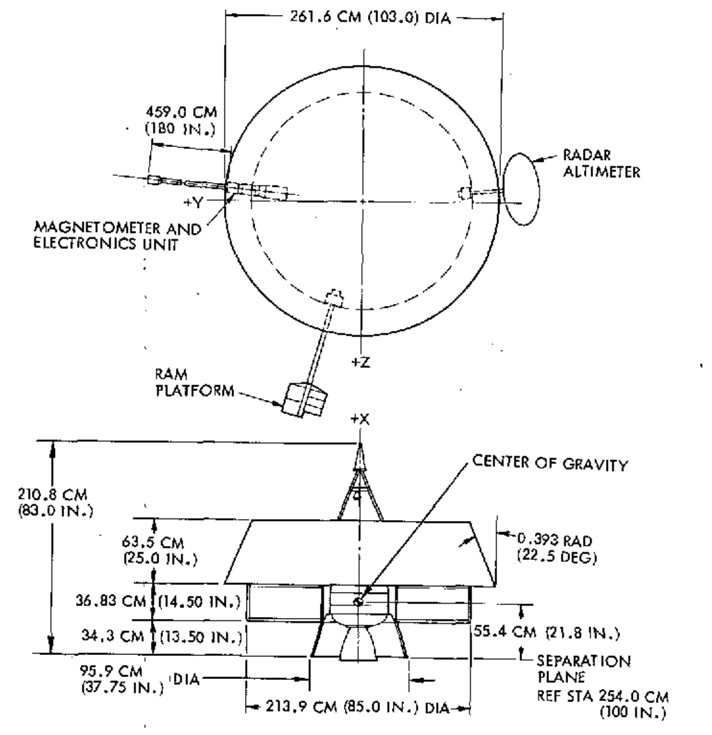


Figure 6-28. Preferred Earth-Pointing Atlas/Centaur Spacecraft Mass Properties Summary, Version IV Science Payload and 1978 Mission Launches

FOLDOUT FRAME 1

FOLDOUT FRAME 2

FOLDOUT FRAME 3

contingency margins exist for both configurations, even for the 1978 launch opportunity, which is inferior to 1977. For the probe and orbiter spacecrafts the congingency margins are 113 kg (250 lb) and 86 kg (190 lb), respectively. As a percentage of dry spacecraft weight, this represents a contingency factor of 17.5 percent for the probe and 33.2 percent for the orbiter; these exceed the preliminary contingency analysis requirements (Appendix 6A) of an estimated 15.8 percent for the probe and 13.2 percent for the orbiter. The science payload weight also includes a weight allowance of 15 percent for present uncertainty.

Spacecraft mass properties characteristics for various flight conditions are summarized in Figure 6-28B and are based on the coordinate reference axes and notation systems presented in Figure 6-28C.

The inertia parameters (λ) for the probe spacecraft before probe separation range in value between 0.41 and 0.42, and during and after probe separation from 0.59 to 0.78. For the orbiter spacecraft the λ value ranges between 0.53 and 0.64. Two dampers are used in the probe spacecraft configuration and a single damper in the orbiter.

All expendables, separable elements, and deployables are located in the composite longitudinal center-of-gravity plane of the spacecraft bus (i.e., in the longitudinal center-of-gravity plane of the spacecraft less the large probe) to minimize principal spin-axis misalignment of the probe spacecraft during periods of asymmetric separation of the small probes and the deployment of scientific sensors before Venus entry. Radial shift of the spin axis occurs during these periods; however, the spin axis remains parallel to the longitudinal reference axis, since products of inertia are not induced in the XY and XZ planes.

Similarly, for the Atlas/Centaur orbiter mission, all expendables, and deployables with the exception of the radar altimeter antenna are located in the composite longitudinal center-of-gravity plane of the spacecraft bus. The radar altimeter antenna is stowed in such a manner that there is no change in the products of inertia about the spacecraft center of gravity in the deployed position. The only problem with this scheme is the possibility of a failure mode where only partial deployment of the antenna causes a degradation in mission performance resulting from a change in products of inertia and associated principal axis shift. A method

A/C IV which would eliminate this problem is to counter-balance the weight of
A/C IV deployable mass in such a manner that even partial deployment does not change the products of inertia. The disadvantage of the counterweight is approximately 2.72 kg (6-lb) increase in spacecraft weight. The most desirable deployment scheme is an in-plane deployment. However, current estimates of dimensions preclude stowing the antenna in the center of gravity.

For both the probe and orbiter spacecraft configurations, the launch vehicle mass properties requirements are less stringent than spacecraft operational considerations. A preliminary allocation of 0.0035 radian (0.20 degree) for the principal axis uncertainty for both the orbiter and the probe spacecraft during probe separation and bus entry has been made to satisfy the error budget for attitude determination and probe deployment.

Figure 6-28B summarizes mass properties and principal axes orientation during various flight conditions.

Optional Despun Reflector Orbiter Configuration for the Version IV Science Payload

A/C IV

A weight summary comparing the optional orbiter configuration with the preferred configuration is presented in Table 6-15. The optional configuration is feasible from a mass properties consideration, having adequate weight contingency and being inertially stable. The detailed weight breakdown for the optional configuration is presented in Appendix 6D.

Table 6-15. Atlas/Centaur Optional Despun Reflector Orbiter Configuration Weight Comparison (Version IV Science Payload)

DESCRIPTION	WEIGHT: EARTH POINTING PREFERRED CONFIGURATION [KG (LB)]		WEIGHT: DESPUN REFLECTOR OPTIONAL CONFIGURATION [KG (LB)]	
ELECTRICAL POWER	39.4	(86.8)	44.5	(98.1)
COMMUNICATIONS	15.0	(33.1)	12.7	(27.9)
ELECTRICAL DISTRIBUTION	15.8	(34.8)	15.8	(34.8)
DATA HANDLING	18.4	(40.5)	18.4	(40.5)
ATTITUDE CONTROL	4.7	(10.3)	12.8	(28.3)
PROPULSION (DRY)	7.9	(17.4)	7.9	(17.4)
SOLID INSERTION MOTOR (BURNOUT)	18.6	(41.1)	18.6	(41.1)
THERMAL CONTROL	15.2	(33.5)	15.0	(33.0)
STRUCTURE	73.7	(162.5)	72.0	(158.8)
BALANCE WEIGHT PROVISION	5.4	(12.0)	5.4	(12.0)
SPACECRAFT BUS LESS SCIENCE (DRY)	214.1	(472.0)	223.1	(491.9)
SCIENTIFIC INSTRUMENTS	45.4	(100.1)	45.4	(100.1)
SPACECRAFT (DRY)	259.5	(572.1)	268.5	(592.0)
INSERTION MOTOR EXPENDABLES	144.5	(318.5)	144.5	(318.5)
HYDRAZINE PROPELLANT AND PRESSURANT	18.1	(39.9)	18.1	(39.9)
SPACECRAFT LESS CONTINGENCY	422.1	(930.5)	431.1	(950.4)
CONTINGENCY (NET ALLOWABLE)	85.9	(189.5)	76.9	(169.6)
(PERCENT OF DRY SPACECRAFT WEIGHT)	(33.1%)		(28.6%)	
GROSS SPACECRAFT AFTER SEPARATION	508.0	(1120.0)	508.0	(1120.0)

Requirements for Version III Science Payload and 1977/1978 Mission Launches

A/C III A/C III

The probe and orbiter spacecraft mass properties requirements imposed by the Atlas/Centaur launch vehicle and mission considerations are summarized in Table 6-16 for the Version III science payload and the 1977/1978 mission launches.

Table 6-16. Atlas/Centaur Spacecraft Mass Properties Requirements for Version III Science Payload and 1977/1978 Mission Launches

PROPERTY	REQUIREMENT	
	PROBE SPACECRAFT	ORBITER SPACECRAFT
SPACECRAFT WEIGHT AFTER SEPARATION	771.1 KG (1700 LB) MAXIMUM (1977, TYPE I) TRAJECTORY	435.4 KG (960 LB) MAXIMUM (1978, TYPE II) TRAJECTORY
LONGITUDINAL CENTER-OF-GRAVITY LIMITATION (DISTANCE OF SPACECRAFT CENTER OF GRAVITY FORWARD OF SPACECRAFT/LAUNCH VEHICLE SEPARATION PLANE)	(NOT A STRINGENT LAUNCH VEHICLE REQUIREMENT)	
RADIAL CENTER OF GRAVITY OFFSET FROM SPACECRAFT CENTERLINE DURING LAUNCH CONDITIONS	(NOT A STRINGENT LAUNCH VEHICLE REQUIREMENT)	
INERTIA RATIO (RATIO OF SPIN TO TRANSVERSE INERTIAS FOR LONG DURATION SPIN-STABILITY CONSIDERATIONS)	>1.10	>1.10
SPACECRAFT PRINCIPAL SPIN AXIS PARALLEL TO SPACECRAFT LONGITUDINAL CENTERLINE DURING LAUNCH CONDITIONS)	(NO SPECIFIC BALANCE REQUIREMENTS ARE STIPULATED BY LAUNCH VEHICLE REQUIREMENTS)	
PRINCIPAL SPIN AXIS IN THE XY AND XZ PLANE PARALLEL TO THE X-AXIS	≤0.0035 RADIAN (≤0.20 DEGREE) (PRELIMINARY ALLOCATION) DURING PROBE SEPARATION AND BUS REENTRY	≤0.0035 RADIAN (≤0.20 DEGREE) (PRELIMINARY ALLOCATION) DURING VENUS ORBIT OPERATIONS

Preferred Spacecraft Configurations for Version III Science Payload and 1977/1978 Mission Launches

A/C III 31 W A/C III

The detailed weight breakdowns for the preferred probe and fanbeam, fanscan orbiter spacecraft configurations for Version III science payload and 1977/1978 mission launches are presented in Figure 6-29A. Even with the earlier estimate of Atlas/Centaur performance, sufficient weight contingency margins exist for both configurations. For the probe and orbiter spacecrafts the contingency margins are 124. 7 kg (275 lb) and 43. 2 kg (95. 2 lb), respectively. As a percentage of dry spacecraft weight this represents a contingency factor of 19. 9 percent for the probe and 17. 1 percent for the orbiter, which exceed preliminary contingency analyses requirements (Appendix 6E) of an estimated 15. 8 percent for the probe and 13. 2 percent for the orbiter.

A DETAILED WEIGHT SUMMARY

DESCRIPTION	PROBE MISSION (KG)	PROBE MISSION (LB)	ORBITER MISSION (KG)	ORBITER MISSION (LB)
ELECTRICAL POWER	20.0	44.0	50.8	112.0
SOLAR ARRAY ASSEMBLY (SIX PANELS)	8.16	18.0	17.46	38.5
BATTERY	1.59	3.5	19.60	43.2
POWER CONTROL UNIT INCLUDING SHUNT	4.45	9.8	6.35	14.0
CTRF/INVERTER	5.76	12.7	7.39	16.3
COMMUNICATIONS	15.5	34.1	12.5	27.5
CONSCAN PROCESSOR	----	----	0.36	0.8
RECEIVERS (2)	4.90	10.8	2.36	5.2
TRANSMITTER DRIVERS (2)	1.27	2.8	1.09	2.4
POWER AMPLIFIERS (4)	----	----	1.81	4.0
TWT	3.62	8.0	----	----
HYBRIDS (1/5)	0.05	0.1	0.23	0.5
DIPLEXERS (2)	1.95	4.3	1.95	4.3
SWITCHES (4/5)	1.09	2.4	1.36	3.0
FORWARD OMNI	0.14	0.3	0.14	0.3
AFT OMNI	0.41	0.9	0.23	0.5
MEDIUM GAIN ANTENNA	0.91	2.0	----	----
FAN-BEAM ANTENNA	----	----	1.13	2.5
FANSCAN ANTENNA	----	----	0.45	1.0
RF COAX AND CONNECTORS	1.13	2.5	1.36	3.0
ELECTRICAL DISTRIBUTION	15.5	34.1	15.8	34.8
COMMAND DISTRIBUTION UNIT	4.13	9.1	4.45	9.8
HARNESS AND CONNECTORS	11.34	25.0	11.34	25.0
DATA HANDLING	3.9	8.5	12.5	27.5
DIGITAL TELEMETRY UNIT	3.08	6.8	3.08	6.8
DIGITAL DECODER UNIT (2)	0.77	1.7	0.77	1.7
DATA STORAGE UNIT (3)	----	----	8.62	19.0
ATTITUDE CONTROL	2.7	6.0	2.7	6.0
CONTROL ELECTRONICS ASSEMBLY	2.31	5.1	2.31	5.1
SUN SENSOR ASSEMBLY (2)	0.41	0.9	0.41	0.9
PROPULSION (DRY)	6.9	15.3	6.9	15.3
PROPELLANT TANK ASSEMBLY (3)	3.13	6.9	3.13	6.9
THRUSTER ASSEMBLY (8)	2.18	4.8	2.18	4.8
FILTER	0.18	0.4	0.18	0.4
PRESSURE TRANSDUCER	0.18	0.4	0.18	0.4
FILL AND DRAIN VALVE ASSEMBLY	0.18	0.4	0.18	0.4
PROPELLANT LINES AND MISCELLANEOUS	1.09	2.4	1.09	2.4
SOLID INSERTION MOTOR (BURNOUT)	----	----	18.7	41.1
THERMAL CONTROL	15.5	34.3	20.2	44.5
INSULATION ASSEMBLY	10.12	22.3	8.12	17.9
FORWARD CLOSURE ASSEMBLY	1.09	2.4	1.32	2.9
SIDE CLOSURE ASSEMBLY	0.54	1.2	0.54	1.2
LOUVER ASSEMBLY (3/7 SQUARE FEET)	1.77	3.9	4.13	9.1
THERMAL FIN-TRANSMITTER	0.68	1.5	4.72	10.4
HEATERS, ISOLATORS, PAINT, ETC.	1.36	3.0	1.36	3.0
STRUCTURE	79.6	175.5	73.4	161.8
CENTRAL CYLINDER ASSEMBLY	20.82	45.9	22.77	50.2
UPPER RING	(3.99)	(8.8)	(2.90)	(6.4)
CYLINDER AND LOWER FRUSTUM	(8.21)	(18.1)	(8.08)	(17.8)
PLATFORM SUPPORT RING	(1.54)	(3.4)	(1.54)	(3.4)
SEPARATION RING	(6.62)	(14.6)	(6.62)	(14.6)
MOTOR MOUNTING RING	----	----	(2.95)	(6.5)
ATTACH HARDWARE	(0.46)	(1.0)	(0.68)	(1.5)

DESCRIPTION	PROBE MISSION (KG)	PROBE MISSION (LB)	ORBITER MISSION (KG)	ORBITER MISSION (LB)
STRUCTURE (CONTINUED)				
PLATFORM/COMPARTMENT ASSEMBLY	27.03	59.6	27.03	59.6
UPPER STRUTS (12)	(2.59)	(5.7)	(2.59)	(5.7)
PLATFORM STRUTS (6)	(1.68)	(3.7)	(1.68)	(3.7)
VERTICALS (6)	(1.81)	(4.0)	(1.81)	(4.0)
UPPER RING ASSEMBLY	(2.54)	(5.6)	(2.54)	(5.6)
PLATFORM STRUT FITTINGS (6)	(0.82)	(1.8)	(0.82)	(1.8)
PLATFORM ASSEMBLY	(15.69)	(34.6)	(15.69)	(34.6)
BRACKETS AND ATTACH HARDWARE	(1.90)	(4.2)	(1.90)	(4.2)
SOLAR ARRAY SUPPORT ASSEMBLY	5.90	13.0	5.81	12.8
UPPER RING	(1.64)	(3.6)	(1.55)	(3.4)
LOWER RING	(1.81)	(4.0)	(1.81)	(4.0)
STRUTS (18)	(2.45)	(5.4)	(2.45)	(5.4)
PROBE SUPPORT AND RELEASE MECHANISMS (4)	8.03	17.7	----	----
ANTENNA SUPPORT ASSEMBLY	----	----	2.90	6.4
MAGNETOMETER BOOM ASSEMBLY	4.22	9.3	4.22	9.3
PROPULSION SUPPORT ASSEMBLY	2.27	5.0	2.27	5.0
DAMPER (2/1)	2.72	6.0	2.72	6.0
FORWARD OMNI SUPPORT	0.23	0.5	----	----
AFT OMNI SUPPORT	0.68	1.5	0.23	0.5
MEDIUM GAIN ANTENNA SUPPORT	0.23	0.5	----	----
SCIENCE SUPPORT BRACKETRY	1.36	3.0	1.36	3.0
NM/IM SPECTROMETER SUPPORTS (2)	2.04	4.5	----	----
EQUIPMENT TIE-DOWN AND INTEGRATION HARDWARE	4.08	9.0	4.08	9.0
BALANCE WEIGHT PROVISION	5.4	12.0	5.4	12.0
• SPACECRAFT BUS LESS SCIENCE (DRY)	165.0	363.8	218.9	482.5
SCIENCE INSTRUMENTS	12.0	26.4	33.0	72.9
NEUTRAL MASS SPECTROMETER	5.44	12.0	5.44	12.0
ION MASS SPECTROMETER	1.45	3.2	1.45	3.2
ELECTRON TEMPERATURE PROBE	1.00	2.2	1.00	2.2
UV FLUORESCENCE	1.59	3.5	----	----
MAGNETOMETER	2.49	5.5	2.49	5.5
UV SPECTROMETER	----	----	5.44	12.0
IR RADIOMETER	----	----	4.54	10.0
RF ALTIMETER	----	----	12.70	28.0
• SPACECRAFT BUS (DRY)	177.0	390.2	251.9	555.4
PROBES (WITHOUT CONTINGENCY)	451.0	994.2	----	----
LARGE PROBE	257.5	567.6	----	----
SMALL PROBES (3)	193.5	426.6	----	----
• SPACECRAFT (DRY)	628.0	1384.4	251.9	555.4
PROPELLANTS AND PRESSURANT	18.4	40.5	140.3	309.4
INSERTION PROPELLANT AND EXPENDED INERTS	----	----	126.10	278.0
HYDRAZINE PROPELLANT	18.10	39.9	13.97	30.8
NITROGEN PRESSURANT	0.27	0.6	0.27	0.6
• SPACECRAFT LESS CONTINGENCY	646.4	1424.9	392.2	864.8
CONTINGENCY (NET ALLOWABLE)	124.7	275.1	43.2	95.2
(PERCENT OF DRY SPACECRAFT WEIGHT)	(19.9)		(17.1)	
• GROSS SPACECRAFT AFTER SEPARATION	771.1	1700.0	435.4	960.0

B SUMMARY OF SPACECRAFT MASS PROPERTIES DURING VARIOUS FLIGHT CONDITIONS

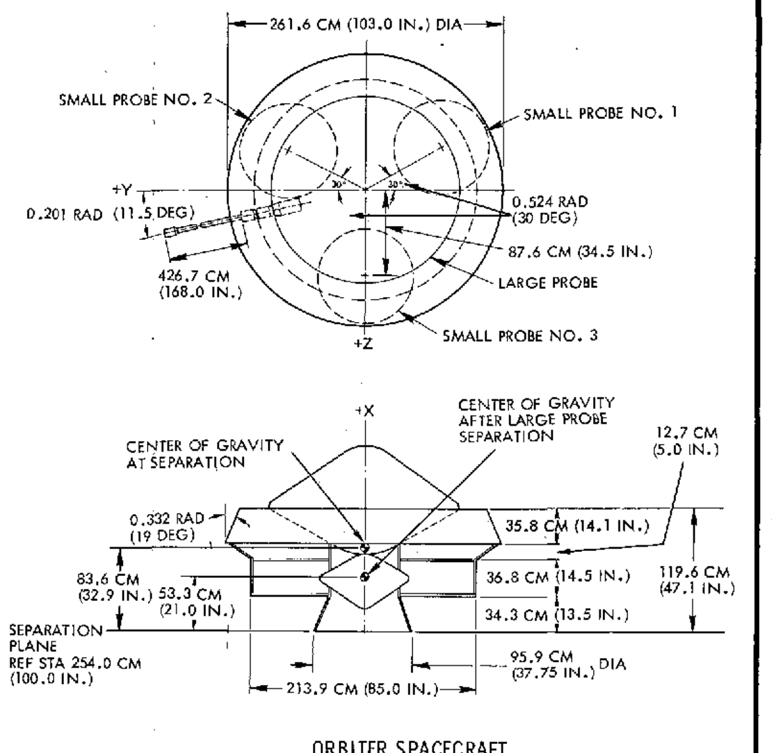
CONDITION	WEIGHT (KG)	WEIGHT (LB)	CG X (CM)	CG X (IN)	CG Y (CM)	CG Y (IN)	CG Z (CM)	CG Z (IN)	I_X (SPIN) (KG-M²)	I_X (SPIN) (SLUG FT²)	I_Y (KG-M²)	I_Y (SLUG FT²)	I_Z (KG-M²)	I_Z (SLUG FT²)	P_{XY} (KG-M²)	P_{XY} (SLUG FT²)	P_{XZ} (KG-M²)	P_{XZ} (SLUG FT²)	P_{YZ} (KG-M²)	P_{YZ} (SLUG FT²)	RATIO(1) I_X/I_Y	RATIO(1) I_X/I_Z	(1)(2) λ, INERTIA PARAMETER
PROBE MISSION																							
AFTER BOOSTER SEPARATION (MAGNETOMETER STOWED)	771.1	1700	337.6	132.9	0	0	0	0	409.3 (409.3)	301.9 (301.9)	341.5 (344.5)	251.9 (254.1)	329.9 (327.0)	243.3 (241.2)	0	0	0	0	6.5	4.8 (-24.0°)	(1.19)	(1.25)	(0.22)
AFTER BOOSTER SEPARATION (MAGNETOMETER DEPLOYED)	771.1	1700	337.6	132.9	1.02	0.40	0.20	0.08	452.2 (452.3)	333.5 (333.6)	343.2 (337.2)	253.1 (248.7)	371.2 (377.0)	273.8 (278.1)	-2.35	-1.73 (1.22°)	-0.47	-0.35	14.1	10.4 (22.5°)	(1.34)	(1.20)	(0.26)
AFTER MIDCOURSE (MAGNETOMETER DEPLOYED)	761.6	1679	337.8	133.0	1.02	0.40	0.20	0.08	449.7 (449.9)	331.7 (331.8)	341.0 (335.2)	251.5 (247.2)	369.2 (375.0)	272.3 (276.6)	-2.37	-1.75 (1.24°)	-0.47	-0.35	14.1	10.4 (22.6°)	(1.34)	(1.20)	(0.26)
PRIOR PROBE SEPARATION (MAGNETOMETER STOWED)	761.6	1679	337.8	133.0	0	0	0	0	406.9 (406.9)	300.1 (300.1)	339.5 (342.3)	250.4 (252.5)	327.8 (325.0)	241.8 (239.7)	0	0	0	0	6.5	4.8 (-24.0°)	(1.19)	(1.25)	(0.22)
AFTER LARGE PROBE SEPARATION (MAGNETOMETER STOWED)	450.0	992	307.3	121.0	0	0	0	0	383.0 (383.0)	282.5 (282.5)	217.9 (220.9)	160.7 (162.9)	206.2 (203.4)	152.1 (150.0)	0	0	0	0	6.5	4.8 (-24.0°)	(1.73)	(1.88)	(0.81)
AFTER FIRST SMALL PROBE SEP. (MAGNETOMETER STOWED)	370.6	817	307.3	121.0	16.00	6.29	9.22	3.63	298.1 (298.1)	219.9 (219.9)	191.2 (201.7)	141.0 (148.8)	143.2 (132.6)	105.6 (97.8)	0	0	0	0	-24.9	-18.4 (23.1°)	(1.48)	(2.25)	(0.77)
AFTER SECOND SMALL PROBE SEP. (MAGNETOMETER STOWED)	291.2	642	307.3	121.0	0	0	23.47	9.24	222.5 (222.5)	164.1 (164.1)	154.6 (155.4)	114.0 (114.6)	99.1 (98.3)	73.1 (72.5)	0	0	0	0	6.5	4.8 (-6.6°)	(1.43)	(2.26)	(0.74)
AFTER THIRD SMALL PROBE SEP. (MAGNETOMETER STOWED)	209.1	461	307.3	121.0	0	0	0	0	165.5 (165.5)	122.1 (122.1)	101.7 (104.5)	75.0 (77.1)	90.0 (87.2)	66.4 (64.3)	0	0	0	0	6.5	4.8 (-24.0°)	(1.58)	(1.90)	(0.72)
END OF LIFE (MAGNETOMETER DEPLOYED)	209.1	461	307.3	121.0	3.71	1.46	0.76	0.30	208.3 (208.3)	153.6 (153.6)	103.3 (97.3)	76.2 (71.8)	131.1 (136.9)	96.7 (101.0)	0	0	0	0	14.1	10.4 (22.6°)	(2.14)	(1.52)	(0.77)
ORBITER MISSION																							
AFTER BOOSTER SEPARATION (MAGNETOMETER STOWED)	435.4	960	306.6	120.7	0	0	0	0	211.8 (211.8)	156.2 (156.2)	131.1 (127.4)	96.7 (94.0)	141.0 (144.7)	104.0 (106.7)	0	0	0	0	-7.2	-5.3 (-27.7°)	(1.66)	(1.46)	(0.55)
AFTER BOOSTER SEPARATION (MAGNETOMETER DEPLOYED)	435.4	960	306.6	120.7	1.78	0.70	0.36	0.14	254.6 (254.6)	187.8 (187.8)	132.7 (132.7)	97.9 (97.9)	182.2 (182.2)	134.4 (134.4)	0	0	0	0	0.5	0.4 (0.6°)	(1.92)	(1.40)	(0.60)
AFTER MIDCOURSE (MAGNETOMETER DEPLOYED)	429.5	947	306.6	120.7	1.80	0.71	0.36	0.14	253.0 (253.0)	186.6 (186.6)	131.9 (131.9)	97.3 (97.3)	181.4 (181.4)	133.8 (133.8)	0	0	0	0	0.5	0.4 (0.6°)	(1.92)	(1.39)	(0.60)
PRIOR VENUS ORBIT INSERTION (MAGNETOMETER STOWED)	429.5	947	306.6	120.7	0	0	0	0	210.1 (210.1)	155.0 (155.0)	130.3 (126.6)	96.1 (93.4)	140.2 (143.9)	103.4 (106.1)	0	0	0	0	-7.2	-5.3 (-27.7°)	(1.66)	(1.46)	(0.55)
AFTER VENUS ORBIT INSERTION (MAGNETOMETER STOWED)	303.5	669	306.6	120.7	0	0	0	0	204.0 (204.0)	150.5 (150.5)	124.2 (120.4)	91.6 (88.8)	134.0 (137.7)	98.8 (101.6)	0	0	0	0	-7.2	-5.3 (-27.7°)	(1.69)	(1.48)	(0.58)
AFTER VENUS ORBIT INSERTION (MAGNETOMETER DEPLOYED)	303.5	669	306.6	120.7	2.57	1.01	0.51	0.20	246.8 (246.8)	182.0 (182.0)	125.7 (125.7)	92.7 (92.7)	175.2 (175.2)	129.2 (129.2)	0	0	0	0	0.5	0.4 (0.6°)	(1.96)	(1.41)	(0.63)
END OF LIFE (MAGNETOMETER DEPLOYED)	295.7	652	306.6	120.7	2.62	1.03	0.53	0.21	244.9 (244.9)	180.6 (180.6)	124.7 (124.7)	92.0 (92.0)	174.2 (174.2)	128.5 (128.5)	0	0	0	0	0.5	0.4 (0.6°)	(1.96)	(1.41)	(0.62)

NOTE: (1) VALUES IN PARENTHESES ARE REFERENCED RELATIVE TO THE PRINCIPAL AXES.

(2) INERTIA PARAMETER, $\lambda = \left[\left(\frac{I_X - I_Y}{I_Y}\right)\left(\frac{I_X - I_Z}{I_Z}\right)\right]^{0.5}$

C COORDINATE REFERENCE AXES AND SYSTEM NOTATION SYSTEM

PROBE SPACECRAFT



ORBITER SPACECRAFT

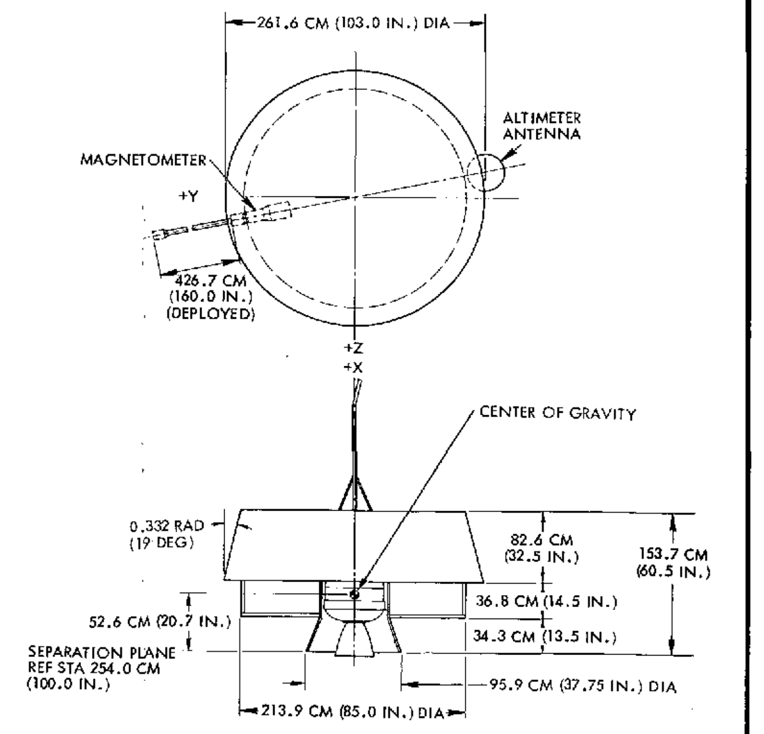


Figure 6-29. Preferred Fanbeam, Fanscan Atlas/Centaur Spacecraft Mass Properties Summary for Version III Science Payload and 1977/1978 Mission Launches

A/C III Spacecraft mass properties characteristics for various flight conditions are summarized in Figure 6-29B and are based on the coordinate reference axes and notation systems presented in Figure 6-29C.

The inertia parameters (λ) for the probe spacecraft before probe separation range in value between 0.22 and 0.26, and during and after probe separation between 0.72 and 0.81. For the orbiter spacecraft the λ value ranges between 0.55 and 0.63.

To minimize principal spin-axis misalignment during asymmetric separation of the small probes and/or deployment of scientific sensors, the same mass properties control philosophy as discussed earlier in this section is followed. A principal spin-axis misalignment of ~0.023 radian (~1.3 degrees) occurs during transit conditions for the probe bus when the magnetometer sensor is deployed. However, during sequenced small probe separation and experience deployment before bus entry into Venus, there is no principal axis tilt. All separations and deployments occur in the longitudinal center-of-gravity plane, eliminating products of inertia.

Similarly, for the Atlas/Centaur orbiter mission, all experiment deployments are in the longitudinal center-of-gravity plane of the spacecraft, eliminating principal axis misalignments caused by sensor deployments.

The mass properties control limits are imposed by the spacecraft operational considerations. A preliminary principal spin-axis uncertainty allocation of 0.0035 radian (0.20 degree) for the orbiter and probe spacecraft during probe separation and bus entry is currently assumed.

Figure 6-29B summarizes details relative to the principal axis orientation.

Optional Orbiter Configurations for Version III Science Payload

A/C III
12 W A/C III
A/C III

A weight summary comparing the three optional orbiter configurations relative to the 36-watt fanbeam, fanscan configuration is presented in Table 6-17. The net allowable contingency for each of the optional configurations exceeds the currently estimated contingency requirement factor of 13.2 percent. As mentioned previously, a preliminary contingency analysis (Appendix 6E) was conducted.

Table 6-17. Atlas/Centaur Optional Orbiter Configuration Weight Comparison Summary, Version III Science Payload.

DESCRIPTION	31 W A/C III		12 W A/C III		A/C III		A/C III	
	[KG (LB)]		[KG (LB)]		[KG (LB)]		[KG (LB)]	
ELECTRICAL POWER	50.8	(112.0)	35.7	(78.6)	35.7	(78.6)	40.8	(89.9)
COMMUNICATIONS	12.5	(27.5)	11.8	(25.9)	15.7	(34.5)	12.7	(27.9)
ELECTRICAL DISTRIBUTION	15.8	(34.8)	15.8	(34.8)	15.8	(34.8)	15.8	(34.8)
DATA HANDLING	12.5	(27.5)	12.5	(27.5)	12.5	(27.5)	12.5	(27.5)
ATTITUDE CONTROL	2.7	(6.0)	2.7	(6.0)	2.7	(6.0)	12.8	(28.3)
PROPULSION (DRY)	6.9	(15.3)	6.9	(15.3)	6.9	(15.3)	6.9	(15.3)
SOLID INSERTION MOTOR (BURNOUT)	18.7	(41.1)	18.7	(41.1)	18.7	(41.1)	18.7	(41.1)
THERMAL CONTROL	20.2	(44.5)	15.0	(33.0)	15.0	(33.0)	15.0	(33.0)
STRUCTURE	73.4	(161.8)	73.0	(161.1)	75.4	(166.4)	72.0	(158.8)
BALANCE WEIGHT PROVISION	5.4	(12.0)	5.4	(12.0)	5.4	(12.0)	5.4	(12.0)
SPACECRAFT BUS LESS SCIENCE (DRY)	218.9	(482.5)	197.5	(435.3)	203.8	(449.2)	212.6	(468.6)
SCIENTIFIC INSTRUMENTS	33.0	(72.9)	33.0	(72.9)	33.0	(72.9)	33.0	(72.9)
SPACECRAFT (DRY)	251.9	(555.4)	230.5	(508.2)	236.8	(522.1)	245.6	(541.5)
INSERTION MOTOR EXPENDABLES	126.1	(278.0)	126.1	(278.0)	126.1	(278.0)	126.1	(278.0)
HYDRAZINE PROPELLANT AND PRESSURANT	14.2	(31.4)	14.2	(31.4)	14.2	(31.4)	14.2	(31.4)
SPACECRAFT LESS CONTINGENCY	392.2	(864.8)	370.8	(817.6)	377.1	(831.5)	385.9	(850.9)
CONTINGENCY (NET ALLOWABLE)	43.2	(95.2)	64.6	(142.4)	58.3	(128.5)	49.5	(109.1)
(PERCENT OF DRY SPACECRAFT WEIGHT)	(17.1%)		(28.0%)		(24.6%)		(20.1%)	
GROSS SPACECRAFT AFTER SEPARATION	435.4	(960.0)	435.4	(960.0)	435.4	(960.0)	435.4	(960.0)

All the optional configurations are spin-stable and the same mass properties control considerations as the preferred configuration apply. The detailed mass properties for the optional configurations are summarized in Appendix 6F.

6.2.1.4 Electrical Design Concept (Atlas/Centaur) A/C IV A/C IV

The preferred electrical design emphasizes:

- Extensive use of existing, flight-proven hardware, particularly components derived from the Pioneer 10 and 11 Program
- Compatibility with either the 26- or 64-meter deep space station networks
- Maximum hardware commonality among the orbiter, probe bus, and probes to minimize overall program costs.

A summary of the major parameters characterizing the preferred Atlas/Centaur configuration is given in Table 6-18. A block diagram for the preferred probe bus spacecraft, depicting the functional relationships between the subsystems, experiments, and probes, is shown in Figure 6-30. The equivalent diagram for the orbiter spacecraft is given in Figure 6-31. More detailed descriptions of the features, characteristics, and performance of the preferred configuration (summarized in this section) are provided in Section 8.

Table 6-18. Atlas/Centaur Preferred Configuration Parameter Summary

PARAMETER	PROBE BUS	ORBITER
SCIENCE PAYLOAD COMPLEMENT		
WEIGHT	12.0 KG	39.5 KG
POWER	21.5 W	89.5 W
SPIN RATE	0.53 RAD/S, NOMINAL 1.05 TO 2.10 RAD/S, PROBE RELEASE	0.53 RAD/S, NOMINAL 6.3 RAD/S ORBIT INSERTION
ELECTRICAL POWER		
SOLAR POWER (NEAR EARTH)	61 W	100 W (160.1 GIGAMETERS)
SOLAR POWER (NEAR VENUS)	111 W	225 W (PEAK)
BATTERY		
TYPE	SILVER-ZINC	NICKEL-CADMIUM
CAPACITY	1.94×10^6 J (540 WH)	1.24×10^6 J (346 WH)
REACTION CONTROL SYSTEM CAPABILITY		
MONOPROPELLANT HYDRAZINE	16.7 KG	15.1 KG
THRUSTERS	4.5 N	4.5 N
ΔV CAPABILITY (MIDCOURSE)	22 M/S	16 M/S
ORBIT INSERTION MOTOR	N/A	AEROJET SVM-2
MAXIMUM THRUST	N/A	21 500 N
DELIVERED IMPULSE	N/A	346 000 N-S
COMMUNICATIONS (TELEMETRY)	PCM/PSK/PM	PCM/PSK/PM
FREQUENCY	2.295 GHZ	2.295 GHZ
RF POWER	6 W	6 W
ANTENNA	1.5-M DISH (28 DBI)	HORN (15.5 DBI)
DATA RATE	1024 BITS/S - BUS ENTRY (64-METER GROUND ANTENNA)	64 BITS/S - END OF MISSION (26-METER GROUND ANTENNA)
COMMUNICATIONS (COMMAND)	PCM/FSK/PM	PCM/FSK/PM
FREQUENCY	2.115 GHZ	2.115 GHZ
BIT RATE	1 BIT/S	1 BIT/S
DATA HANDLING		
DATA RATES	8 TO 1024 BITS/S	8 TO 1024 BITS/S
DATA STORAGE	NONE	1.23 MEGABITS
ATTITUDE CONTROL		
POINTING ACCURACY	<0.017 RAD (< 1 DEG)	<0.017 RAD (<1 DEG)
ATTITUDE DETERMINATION	SUN ASPECT SENSOR, DOPPLER MODULATION/SHIFT	CONSCAN, SUN ASPECT SENSOR, DOPPLER MODULATION/SHIFT
SPACECRAFT WEIGHT	782 KG (INCLUDING PROBES)	510 KG

Electrical Power

A 0.393-radian(22.5-degree) conical solar array was selected to provide a relatively constant power output with varying sun aspect angles (see Figure 6-32) over the forward hemisphere of the spacecraft for both missions. This design concept offers operational flexibility by permitting leisurely execution of midcourse maneuvers and the probe release sequence independent of battery capacity. A net weight savings is realized for the probe bus because a smaller (lighter weight) battery is adequate, and because the conical array allows solar heating of the large probe, eliminating the need for heaters and attendant increased solar array area. A small weight penalty is incurred for the orbiter because of the taller array, but this is more than compensated for by the high-gain antenna weight savings associated with an earth-pointer.

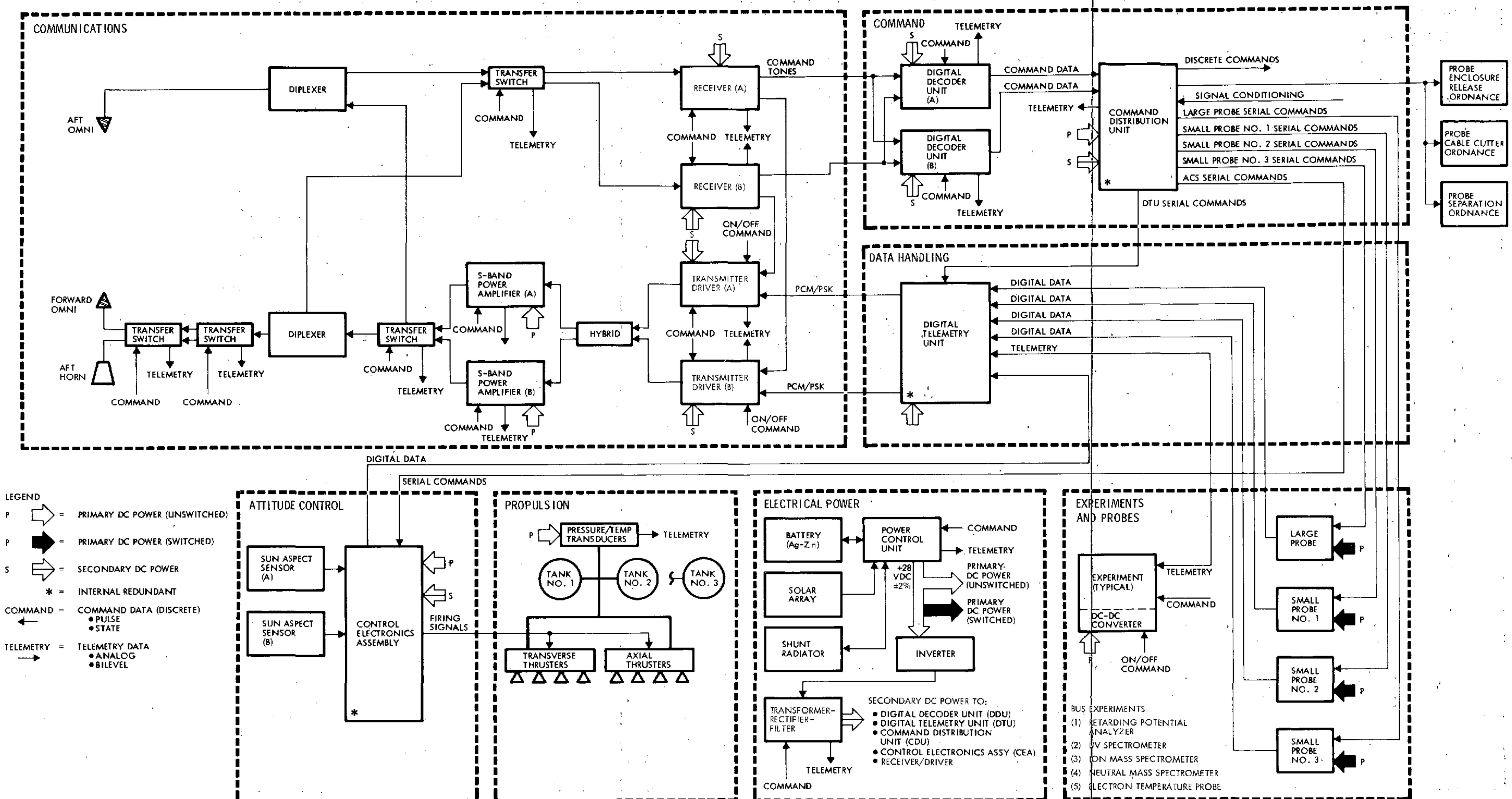


Figure 6-30. Probe Bus Functional Block Diagram, Preferred Atlas/Centaur Configuration

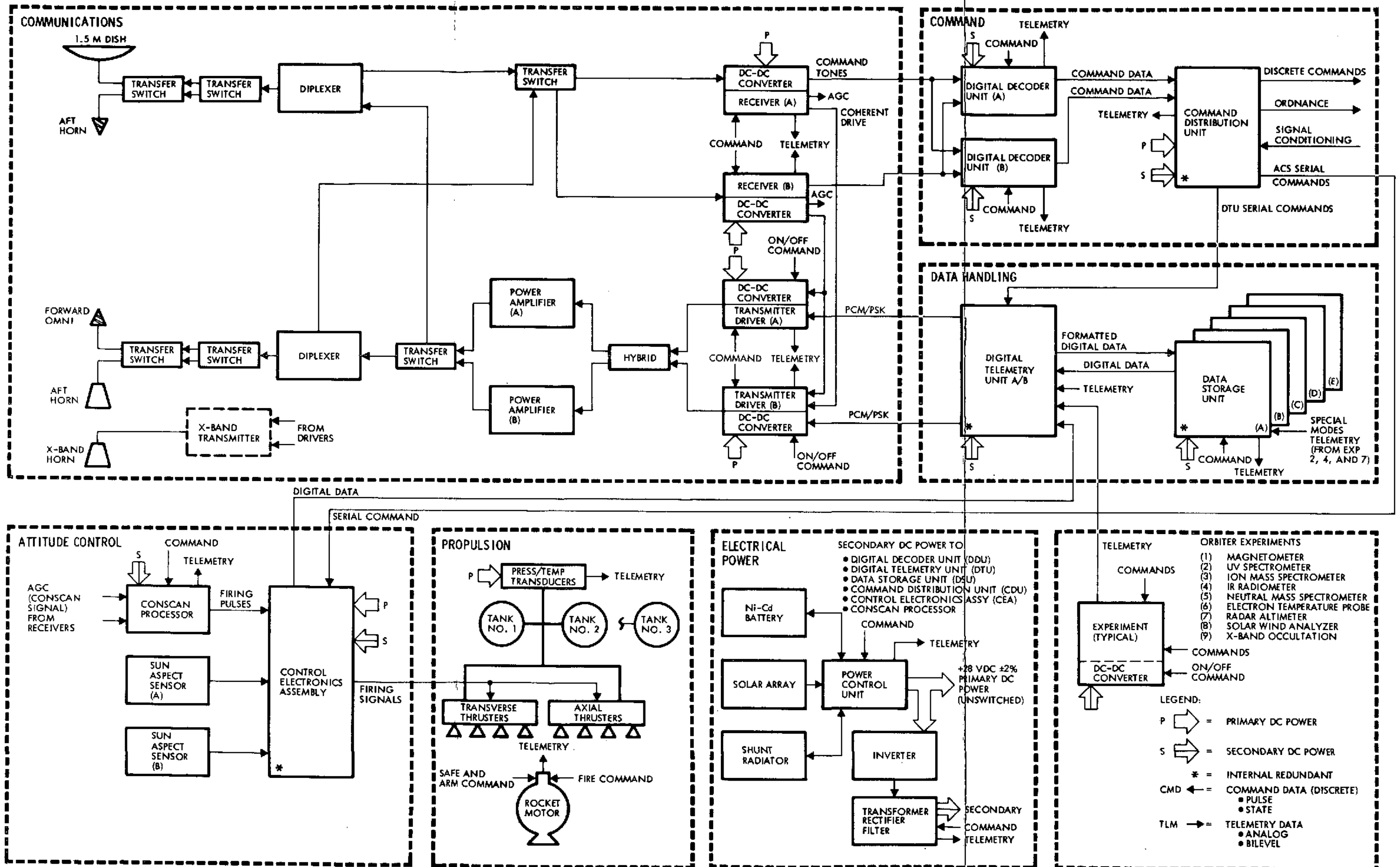


Figure 6-31. Orbiter Functional Block Diagram, Preferred Atlas/Centaur Configuration

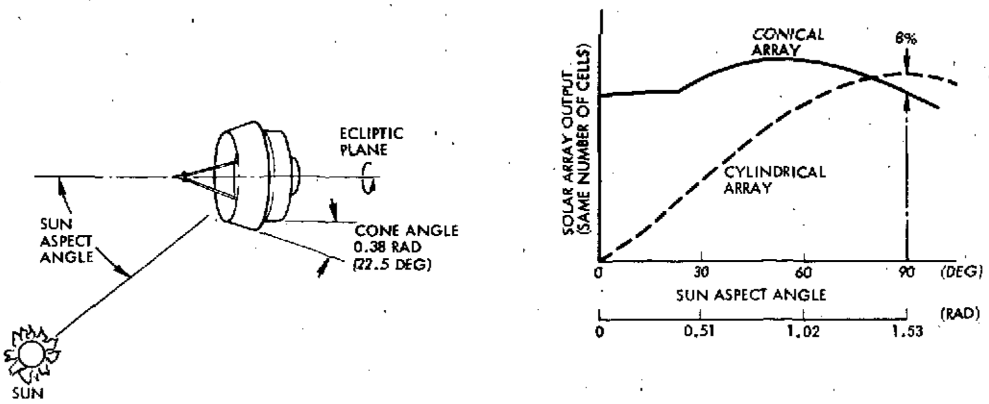


Figure 6-32 Conical Solar Array Characteristics

A silver-zinc (Ag-Zn) battery is recommended for the probe bus because of its superior energy density characteristic and the limited discharge requirements (principally launch loads) of the probe mission. Moreover, the proposed battery cells are identical to those recommended for use in the small probe batteries, thereby maximizing commonality.

In addition to launch loads, the battery requirements for the orbiter mission include supporting spacecraft loads during periodic solar eclipses during the orbit phase of the mission. Nickel-cadmium (Ni-Cd) batteries are characterized by predictable cycle life at high depths of discharge, low cost, and flight-proven experience. Coupled with the relaxed Pioneer Venus magnetic requirements (compared to Pioneers 10 and 11), a Ni-Cd battery was selected for the orbiter.

Power control is achieved using the bus voltage control technique proven on Pioneers 10 and 11. The power control unit (PCU), inverters, and central transformer-rectifier-filter assemblies derived from Pioneers 10 and 11 provide the most cost-effective implementation. Modest modifications, including battery charge/discharge controls, bus filter/telemetry/command slices, and adaptation to 28 VDC input, are required to adapt the existing designs to the Pioneer Venus requirements. A detailed discussion of these modifications is provided in Section 8.1.4.4.

Communications

The probe bus achieves near-spherical uplink and downlink coverage throughout launch, cruise, and midcourse maneuvers via forward and aft log conical spiral antennas. The aft horn antenna is used during the latter

portion of the cruise phase when the bus is earth-pointing and also during atmosphere entry where greater link gain is required to accommodate the high doppler and doppler rates. Figure 6-33 depicts the approximate antenna pattern coverage provided by this antenna arrangement.

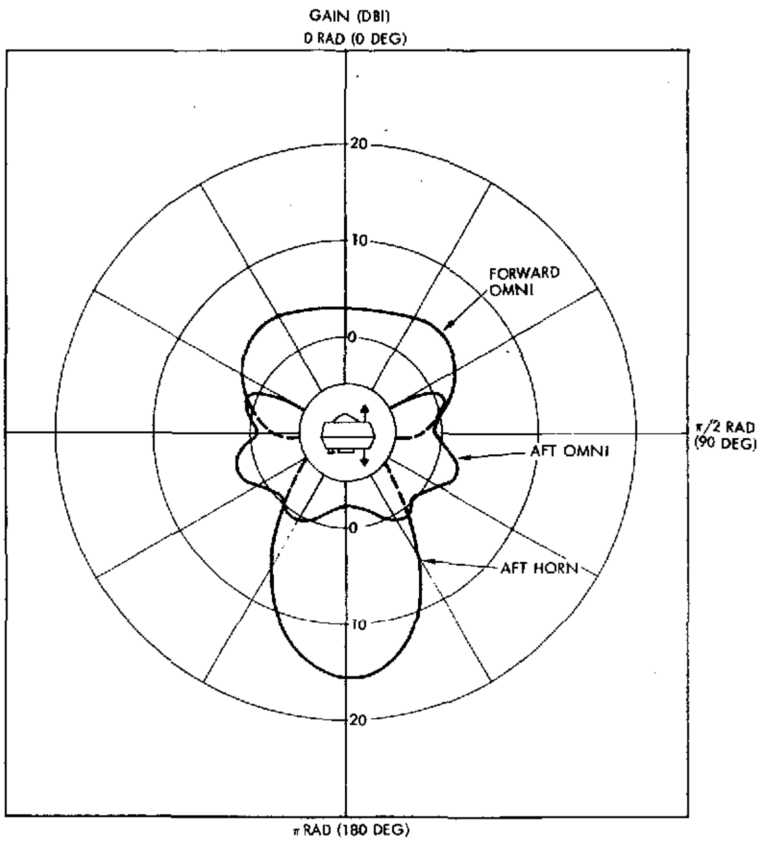


Figure 6-33. Probe Bus Antenna Pattern Coverage

Standby redundant solid-state power amplifiers, coupled to the appropriate antenna, provide sufficient downlink effective isotropic radiated power (EIRP) to support routine tracking and telemetry data acquisition operations with the 26-meter deep space station antenna.

Cross-strapped receivers and transmitter drivers provide reliable two-way coherent doppler for precision tracking, incorporating the fail-safe provisions of the Pioneer 10 and 11 design. The residual units (prototypes, spare, qualification) from the Pioneer 10 and 11 Program offer a considerable cost savings to Pioneer Venus, if made available, and are

compatible with the probe bus configuration; the orbiter requires an X-band coherent drive output, not available on the existing receiver. The Viking transponder is a qualified assembly which closely matches the requirements for this application and is our orbiter baseline selection as well as for the probe bus if Pioneer 10 and 11 equipment is not made available. Maximum commonality is achieved by equipping the probes with the same transponder (small probes use driver only).

In addition to the three antennas discussed above for the probe bus, the preferred orbiter configuration includes a 1.5-meter dish to provide high-rate telemetry data at extended ranges and an X-band horn for use with the occultation experiment. The S-band horn is used for all TT&C operations when the spacecraft aft end is facing earth (approximately L + 108 days to L + 237 days) and supports the dual-frequency occultation experiment. Communications during the orbit insertion maneuver use the forward omni in conjunction with the 64-meter ground station antenna. Figure 6-34 illustrates the approximate antenna pattern coverage afforded by this antenna complement.

The remainder of the communication subsystem configuration is identical to that recommended for the probe bus with the exception of the switching/diplexing network required to interconnect the additional S-band antenna.

The six-watt power amplifier, coupled with the 1.5-meter dish, provides adequate downlink EIRP to support a data rate of 64 bits/s at the maximum range [254.32 gigameters (1.7 AU)] with a 26-meter deep space station. This rate permits reading out all the stored data within a period of 10 hours. Figure 6-35 graphically depicts the telemetry rate capability overlayed on a plot of the orbiter mission trajectory.

Data Handling and Command

The Pioneer 10 and 11 digital telemetry unit (DTU) has been selected for both the probe bus and orbiter applications because it fulfills or surpasses the Pioneer Venus requirements with minor modifications. These modifications are summarized as follows:

- Incorporation of a two-level modulation output (selectable by ground command) to permit downlink modulation index optimization

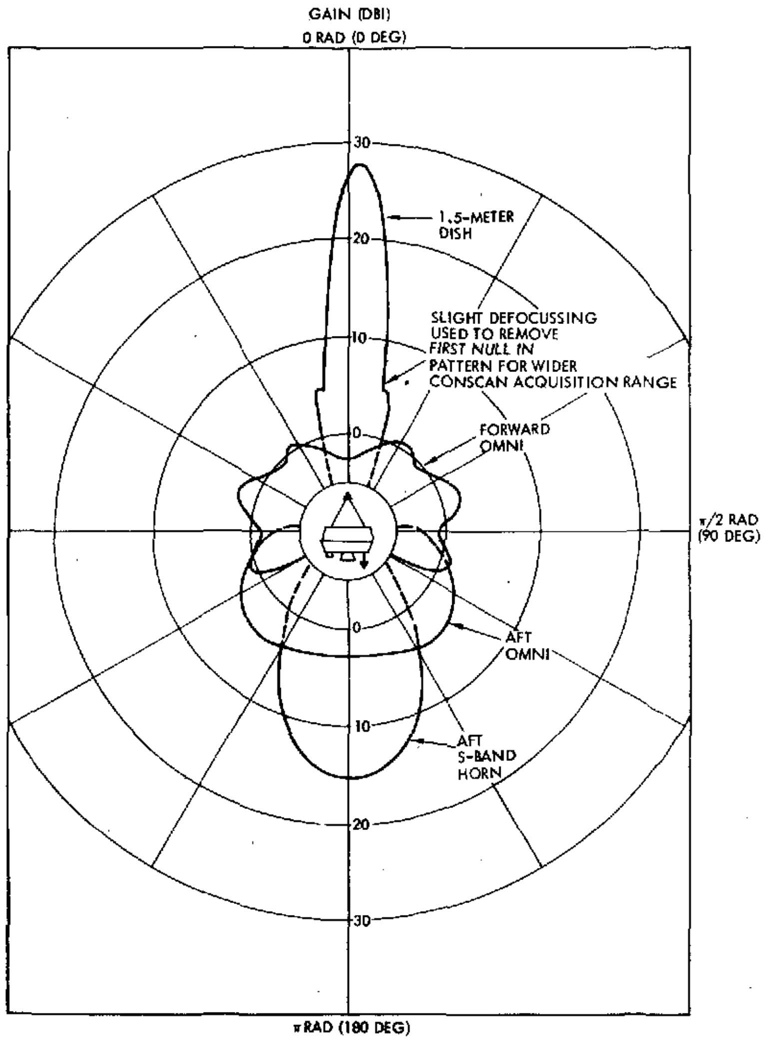


Figure 6-34. Orbiter Antenna Pattern Coverage

- Reduction of lowest available bit rate from 16 to 8 bits/s
- Provision for 10-bit resolution for A/D conversion of analog scientific housekeeping data
- Increase frame length from 192 to 768 bits/s
- Accommodation of analog inputs in the main frame
- Increase number of science formats from two to four.

Storage of instrument data is required only on the orbiter. The selected C-MOS solid-state memory, with a capacity of 1.23 million bits, is comprised of five data storage units (DSU), each containing two modules of 122 880 bits each. The storage capability is required throughout the orbit phase of the mission to store data for delayed transmission when the

THE PIONEER VENUS ORBITER TRAJECTORY IS SHOWN ON AN EARTH-CENTERED COORDINATE SYSTEM WITH THE EARTH-SUN LINE FIXED. RELATIVE POSITION OF THE VEHICLE WITH RESPECT TO VENUS IS SHOWN IN 20-DAY INCREMENTS OVER THE 425-DAY MISSION.

COMMUNICATION RANGE IS MEASURED RADIALLY FROM EARTH ON A LINEAR SCALE. THE CONCENTRIC CIRCLES CENTERED ON EARTH INDICATE THE MAXIMUM TELEMETRY BIT RATE CAPABILITY AT THAT RANGE, ASSUMING DATA ACQUISITION WITH A 26-METER DEEP SPACE STATION. THE 26-METER DEEP SPACE STATION NETWORK CAN SUPPORT THE MAXIMUM SPACECRAFT BIT RATE (64 BITS/S) THROUGHOUT THE MISSION.

TELEMETRY RATES IDENTIFIED WITH AN ASTERISK ARE ACHIEVED WITH THE HORN ANTENNA DURING THE PERIOD L+108 TO L+237 DAYS WHEN THE AFT END OF THE SPACECRAFT FACES EARTH. THE PARABOLIC REFLECTOR ANTENNA IS USED DURING ALL OTHER PHASES OF THE MISSION.

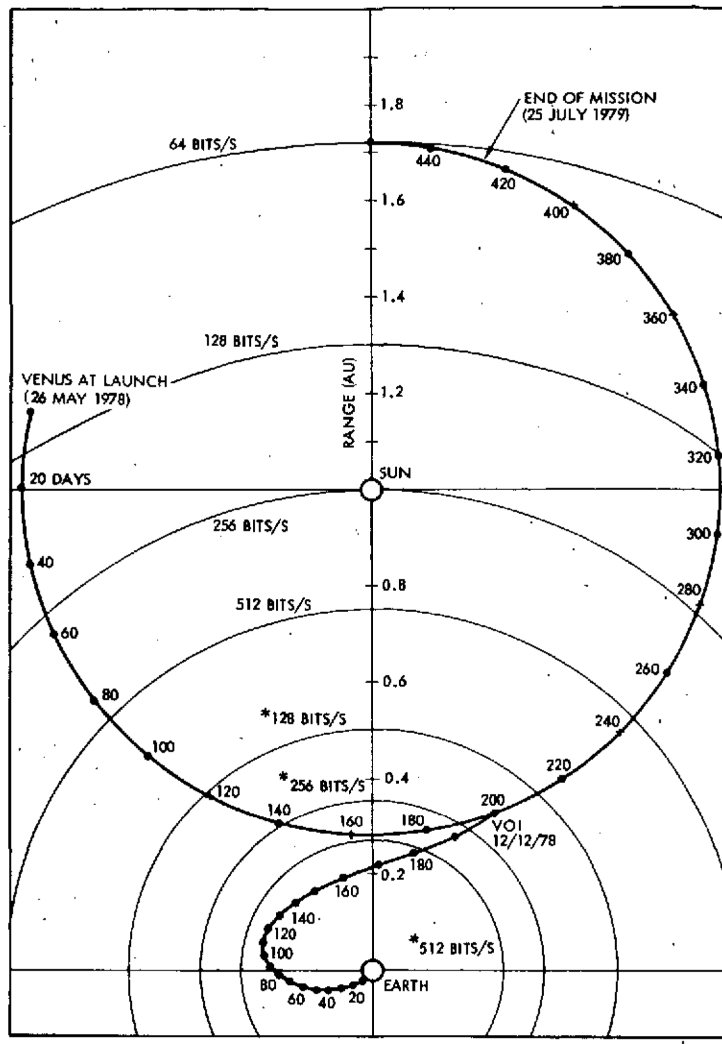


Figure 6-35. Heliocentric Plan View of Pioneer Venus Orbiter Trajectory Showing Telemetry Bit Rate Capability

spacecraft is occulted by Venus and to buffer scientific data acquired at rates exceeding the prevailing telemetry capability. This in-line function, therefore, requires redundancy. In the event of a DSU failure the system can be reconfigured, by ground command, to access the desired input/output data. Conventional types of data storage do not permit the input/output flexibility offered by this design concept.

The command memory function, providing the spacecraft with the capability for storing commands for execution at a later time, has been expanded from that provided in Pioneers 10 and 11. The command memory in the command distribution unit (CDU) has been increased from five stored commands and their associated time delays to a dual system with a capacity of 16 commands. Moreover, the resolution of each incremented time

delay has been significantly increased. The proposed design has 2 seconds uncertainty after an elapsed period of 36 hours, well in excess of the 10-second accuracy required to satisfy orbit insertion timing precision.

This discrete command storage feature simplifies and increases flexibility of other ground operating procedures such as the probe release sequence and instrument operating mode sequencing during routing orbiter data acquisition.

Attitude Determination and Control

The preferred attitude determination and control subsystem (ADCS) configuration uses equipment developed for the Pioneer 10 and 11 and Intelsat III programs, with minimum modifications. The Pioneer 10 and 11 control electronics assembly (CEA) is directly applicable to all Pioneer Venus mission functions and centralizes all ADCS subsystem interfaces, thus providing maximum commonality with other subsystems and components derived from Pioneer designs. Modifications required include 1) the deletion of star sensor logic, and 2) the addition of sun sensor electronics, stored command capabilities, and drivers for additional thrusters.

Attitude determination for both missions is obtained via a combination of earth aspect angle and sun aspect angle measurement. A third measurement provides roll reference. The Intelsat III sun sensor is recommended to provide both a roll reference and a measurement of the sun-spin axis aspect angle. The only required modification to this unit is a change to a V-type slit geometry, accomplished by a simple change in the mask used.

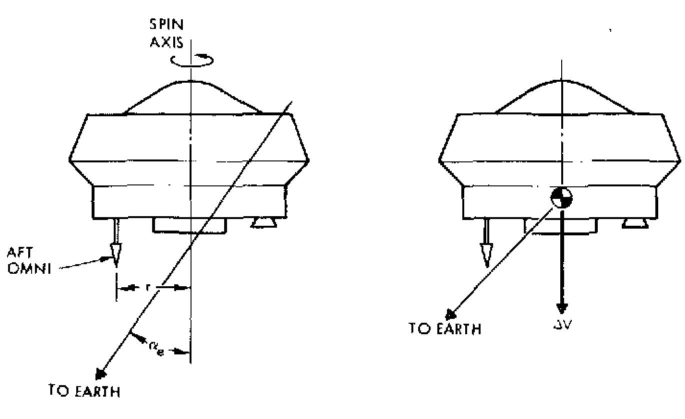


Figure 6-36. Doppler Modulation/Shift Tehcnique for Attitude Determination

The doppler modulation/shift method, successfully used on Pioneers 10 and 11, is the preferred method for probe bus earth aspect angle estimation. Figure 6-36 illustrates the concept. The earth aspect angle can be determined from changes to the signal frequency induced by an offset antenna on the spinning spacecraft. Attitude

determination accuracy is a function of sun-spacecraft-earth geometry, since uniqueness is lost when the sun and earth vectors coincide. For the probe mission, the sun-spacecraft-earth geometry allows attitude determination to accuracies well within requirements for all critical events. Periodic attitude adjustments to maintain illumination of the earth with the horn antenna may be performed using open-loop precession maneuvers.

The orbiter attitude determination is obtained by a combination of the doppler modulation/shift method and the conscan concept used on Pioneers 10 and 11. The latter is based on the amplitude modulation of the uplink RF signal produced by a pointing error when the antenna boresight is tilted with respect to the spin axis. A portion of the antenna pattern is emphasized in Figure 6-37 to represent the range of antenna gain swept through during a spin cycle as a consequence of the pointing error, α_e. Conscan provides a simple space-proven technique for determining and maintaining accurate earth pointing and is far less complex (and costly) than a star mapper.

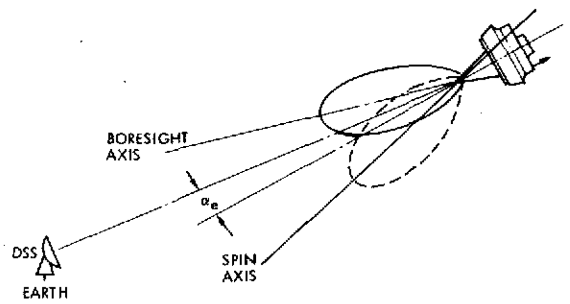


Figure 6-37. Conscan Method for Attitude Determination

The principal advantages of the conscan approach are its good attitude determination accuracy and the operational simplicity attained when it is used for automatic earth-pointing precession maneuvers. Its application to the probe bus, however, is not justified for the modest accuracy improvement obtainable.

Propulsion

The propulsion subsystem consists of the reaction control system (RCS) and the orbit insertion motor (orbiter only). The minimal cost approach emphasizes use of existing hardware which has been previously qualified to levels exceeding Pioneer Venus mission requirements. No development costs and only minimal testing are required.

The RCS uses a monopropellant hydrazine blowdown pressurization scheme similar to Pioneers 10 and 11. Hydrazine propellant is the logical choice since other systems require excessive electrical power (heated

ammonia) or are far too heavy (cold nitrogen) for the required mission specific impulse. The blowdown system is the lightest, most reliable, and least costly system. Three propellant tanks derived from the DSCS-II Program are cost-effective contenders suitable for both missions while providing some margin for growth. Blowdown pressurization and centrifugal force for propellant positioning and expulsion eliminate the need for a bladder and the associated regulators and pressurization components. Common pressurant and propellant lines ensure equal propellant in the three tanks to maintain spacecraft balance.

Eight thrusters (four axial for large ΔV's) are proposed to ensure mission success in the event of a single thruster failure and to avoid coning angle amplification during ΔV maneuvers. The selected thruster is being developed for the FLTSATCOM Program from a dual, isotope-heated unit used on Pioneers 10 and 11. The only modifications are the mechanical coupling of the valve and decomposition chamber and the removal of the isotope heating elements. Moreover, hydrazine thruster impulse repeatability, following initial calibration, is proven and accurate (<3 percent).

The recommended orbit insertion motor is the Aerojet solid-propellant SVM-2 which has been previously used on the Intelsat III Program. A 10 percent off-loading of the design propellant load meets the Pioneer Venus deboost requirements without redesign, requalification, or other critical limitations. This selection affords a more reliable, less expensive, although somewhat heavier, system approach than can be obtained from existing bipropellant systems such as ESRO Symphonie or Mariner '71.

Thermal Control

The major thermal control subsystem features include a developed louver assembly, passive thermal coatings, and multilayer insulation blankets to maintain components within temperature ranges to which they have previously been qualified. The equipment compartment is insulated from the extreme heat influx with aluminized mylar and kapton blankets. Adequate warmth is provided by dissipation of electrical power by electronic components within the compartment; louvers regulate the release of this heat below the mounting platform, maintaining temperatures in the vicinity of the spacecraft equipment and scientific instruments within operating limits.

Dual thermostatically controlled heaters for the thruster valves and catalytic beds, with backup ground commands, prevent the hydrazine from freezing. The equipment compartment is maintained above 3°C (40°F) to eliminate propellant line and tank heaters.

The probe bus solar aspect angle is controlled to maintain acceptable temperatures on the large probe, thereby avoiding the need for a probe heater and controls, jettisonable cover, or both, with the attendant impact on weight, power, and dynamics. With this approach, the internal probe temperature exceeds 15°C (60°F), the desirable lower battery operating temperature at the time of probe release. Insulated jettisonable panels around each of the small probes maintain acceptable probe temperatures during transit.

Aluminized kapton insulation is used in the orbiter to maintain acceptable solid rocket motor compartment temperatures during transit. A jettisonable insulated motor nozzle cap keeps the motor within acceptable temperatures before firing.

Electrical Distribution and Grounding

Figures 6-38 and 6-39 show the electrical distribution diagrams for the probe bus and the orbiter, respectively. The system uses fuses between the power source and loads for primary DC power fault isolation. Secondary DC power fault isolation is achieved using current limiting (150 percent of nominal load) on the DC outputs of the CTRF, as was done on Pioneers 10 and 11.

Equipment Derivation/Status

Table 6-19 summarizes the equipment complement (unit level) for the probe bus and orbiter spacecraft. The subsystem elements are identified by the categories of 1) flight-proven, 2) developed, or 3) new. Their previous program derivation is also given.

Two significant factors are noted. First, extensive commonality between probe bus and orbiter equipment is achieved, as evidenced by the multiple entries in the applications column. Secondly, the majority of equipment is flight-proven and requires only minor modifications for the Pioneer Venus Program. Equally important is the fact that the 1.5-meter dish, silver-zinc battery, and data storage unit, while identified as new designs, are all based on technology well within the present state of the art.

Page intentionally left blank

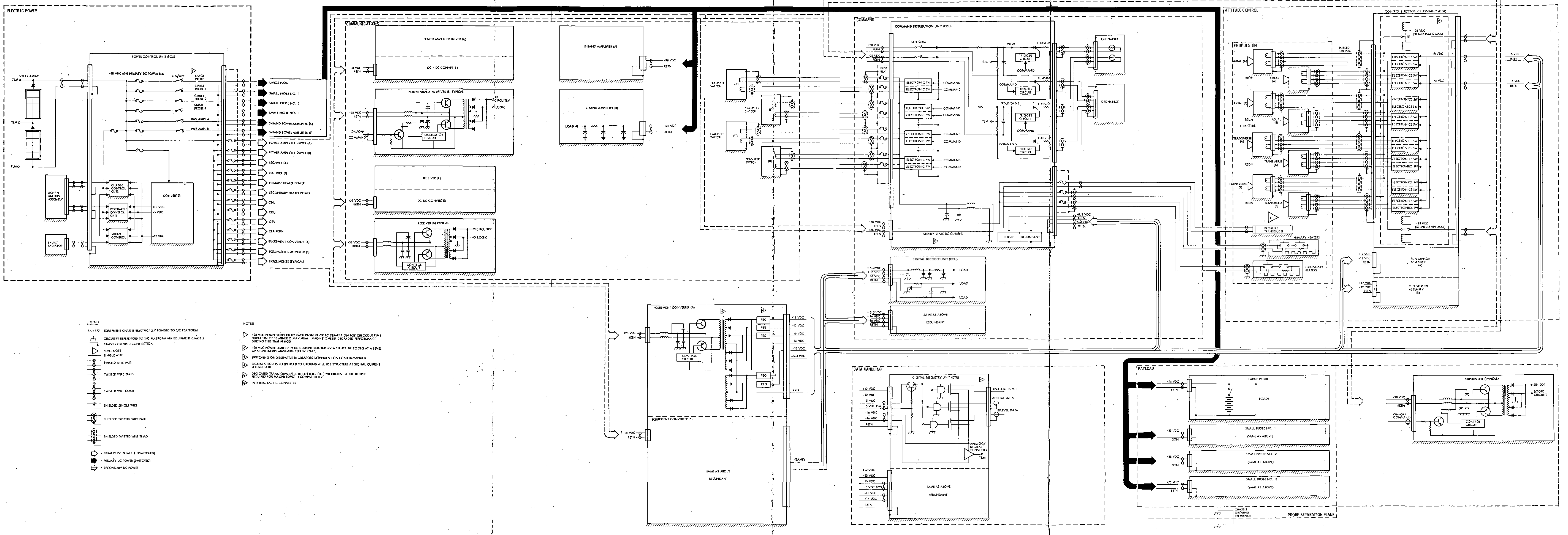


Figure 6-38. Atlas/Centaur Probe Bus Electrical Distribution Diagram

FOLDOUT FRAME 1

FOLDOUT FRAME 2

FOLDOUT FRAME 3

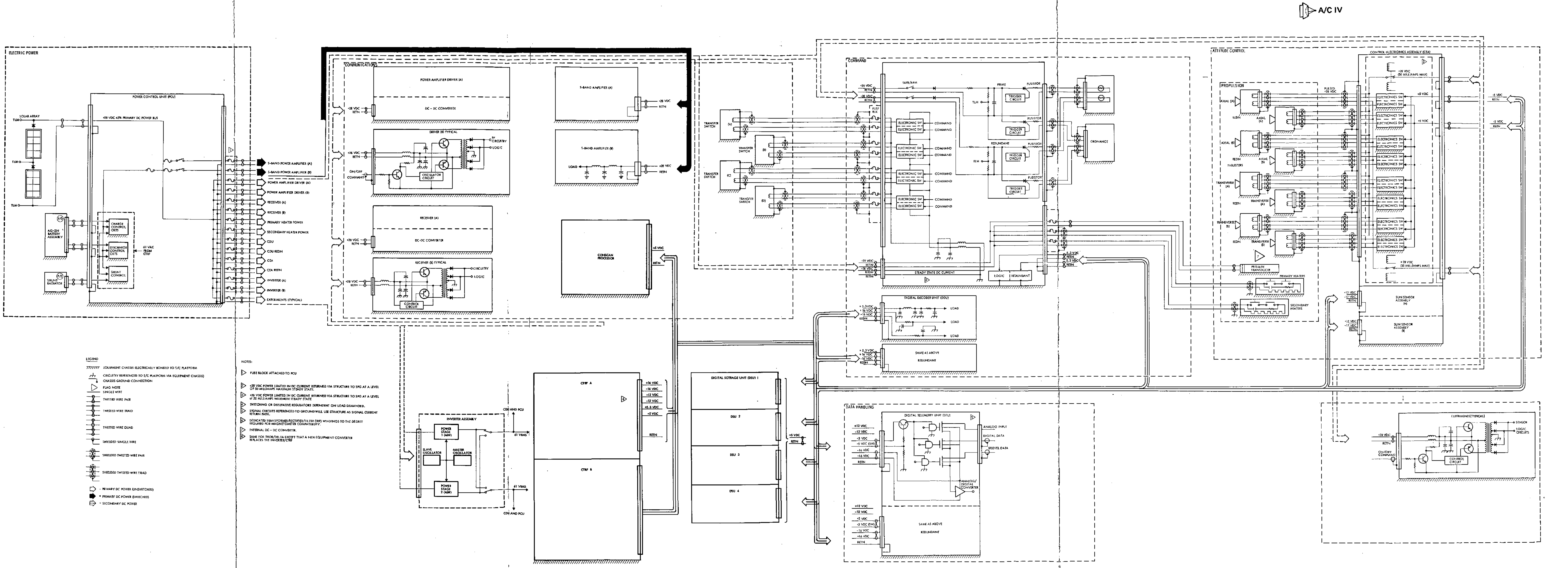


Figure 6-39. Atlas/Centaur Orbiter Electrical Distribution Diagram

Table 6-19. Equipment Derivation/Status, Preferred Atlas/Centaur Configuration

SUBSYSTEM/COMPONENT	APPLICATION PROBE BUS	APPLICATION ORBITER	MANUFACTURER	STATUS*	PREVIOUS USAGE	MODIFICATIONS REQUIRED
COMMUNICATIONS						
TRANSPONDER (RECEIVER-DRIVER)		x	PHILCO-FORD	2	VIKING, SKYNET	INCLUDE CONSCAN AGC; MINOR INTERFACE MODIFICATIONS
TRANSPONDER (RECEIVER-DRIVER)	x		TRW	1	PIONEERS 10 AND 11	NONE
POWER AMPLIFIER (SOLID STATE)	x	x	MICROWAVE SEMICONDUCTOR CORP.	2	COMMERCIAL APPLICATIONS	POSSIBLE GAIN MODIFICATION; INCLUSION OF HIGH RELIABILITY AND SPACE QUALIFICATION REQUIREMENTS
DIPLEXER	x	x	WAVECOM	1	PIONEERS 10 AND 11	NONE
TRANSFER SWITCH	x	x	TELEDYNE	1	PIONEERS 10 AND 11	NONE
HYBRID COUPLER	x	x	ANAREN	2	NONE	SCALE FROM EXISTING L-BAND TO S-BAND
1.5-METER DISH		x	TRW	3	NONE	NOT APPLICABLE
S-BAND MEDIUM-GAIN HORN	x	x	TRW	1	PIONEERS 10 AND 11	NONE
FORWARD OMNI	x	x	TRW	1	PIONEERS 10 AND 11	NONE
X-BAND HORN (OCCULTATION EXPERIMENT)	x	x	TRW	3	NONE	NOT APPLICABLE
DATA HANDLING						
DIGITAL TELEMETRY UNIT	x	x	TRW	1	PIONEERS 10 AND 11	INCREASE MAIN FRAME LENGTH; INCREASE A/D QUANTIZATION; REDUCE LOWEST BIT RATE FROM 16 TO 8 BITS/S; PROVIDE GROUND-COMMANDED TWO-LEVEL MODULATION INDEX CONTROL
DATA STORAGE UNIT		x	TRW	3	NONE	NEW DEVELOPMENT USING EXISTING TECHNOLOGY
COMMAND						
DIGITAL DECODER UNIT	x	x	TRW	1	PIONEERS 10 AND 11	NONE

*1 = FLIGHT-PROVEN; 2 = DEVELOPED; 3 = NEW

SUBSYSTEM/COMPONENT	APPLICATION PROBE BUS	APPLICATION ORBITER	MANUFACTURER	STATUS*	PREVIOUS USAGE	MODIFICATIONS REQUIRED
COMMAND DISTRIBUTION UNIT	x	x	TRW	1	PIONEERS 10 AND 11	AUGMENT ORDNANCE FIRING CIRCUITS; INCREASE COMMAND STORAGE CAPABILITY; ADD ORBIT INSERTION MOTOR FIRING LOGIC
THERMAL						
LOUVERS	x	x	TRW	2	HELIOS	SIX-BLADE ASSEMBLY, NONE; FOUR-BLADE ASSEMBLY, LOWER ASSEMBLY SHORTENED
INSULATION	x	x	TRW	2	PIONEERS 10 AND 11	DESIGNED FOR SPECIFIC APPLICATION
HEATERS	x	x	ELECTRO-FILM	1	PROGRAM 169	NONE
ELECTRICAL POWER						
SOLAR ARRAY	x	x	TRW	2	DEFENSE SUPPORT PROGRAM	SIMILAR TO DEFENSE SUPPORT PROGRAM WITH DIFFERENT CONE ANGLE
BATTERY		x	TRW	2	DEFENSE SUPPORT PROGRAM, DSCS-II	NEW DESIGN BASED ON DEFENSE SUPPORT PROGRAM AND DSCS-II Ni-Cd TECHNOLOGY
BATTERY	x		EAGLE PICHER	3	MMC IRAD	Ag-Zn NEW DESIGN; SAME CELLS AS FOR PROBES
POWER CONTROL UNIT	x	x	TRW	2	PIONEERS 10 AND 11	REDESIGN CHARGE/DISCHARGE CONTROLS; MODIFY BUS FILTER/TELEMETRY/COMMAND SLICES
SHUNT RADIATOR	x	x	THERMAL SYSTEM INC.	1	PIONEERS 10 AND 11	NONE
INVERTER	x	x	TRW	2	PIONEERS 10 AND 11	MODIFY FOR 28 VDC INPUT; ADD REDUNDANT OSCILLATOR
TRANSFORMER-RECTIFIER-FILTER	x	x	TRW	2	PIONEERS 10 AND 11	DELETE RECEIVER AND DRIVER SLICES; ADD DSU SLICE FOR ORBITER ONLY; DELETE CONSCAN PROCESSOR ON PROBE BUS ONLY

*1 = FLIGHT-PROVEN; 2 = DEVELOPED; 3 = NEW

Table 6-19. Equipment Derivation/Status, Preferred Atlas/Centaur Configuration (Continued)

A/C IV

A/C IV

SUBSYSTEM/COMPONENT	APPLICATION PROBE BUS	APPLICATION ORBITER	MANUFACTURER	STATUS*	PREVIOUS USAGE	MODIFICATIONS REQUIRED
PROPULSION						
PROPELLANT TANKS	x	x	PRESSURE SYSTEMS INC.	1	DSCS-II	DELETE BLADDER; MODIFY MOUNTING ARRANGEMENT
THRUSTERS	x	x	TRW		FLTSATCOM	NONE
PRESSURE TRANSDUCER	x	x	STATHAM INSTRUMENTS	1	PIONEERS 10 AND 11	NONE
TEMPERATURE TRANSDUCER	x	x	ROSEMONT ENGINEERING	1	PIONEERS 10 AND 11	NONE
ROCKET MOTOR (ORBITER ONLY)		x	AEROJET SVM-2	1	INTELSAT III	10.2 PERCENT OFF-LOADING
ATTITUDE CONTROL						
CONTROL ELECTRONICS ASSEMBLY	x	x	TRW	1	PIONEERS 10 AND 11	STELLAR REFERENCE ASSEMBLY LOGIC DELETED; ADD REDUNDANT SUN SENSOR LOGIC; ADD VALVE DRIVER LOGIC; ADD SMALL PROBE RELEASE LOGIC
SUN ASPECT SENSOR	x	x	TRW	1	INTELSAT III	MODIFY MASK FIELD OF VIEW FOR IMPROVED ASPECT AND ROLL ACCURACY
CONSCAN PROCESSOR		x	TRW	1	PIONEERS 10 AND 11	IDENTICAL TO PIONEERS 10 AND 11 CONSCAN PROCESSOR EXCEPT CONFIGURE TO ISSUE FIRING PULSES EVERY TWO SPACECRAFT REVOLUTIONS (JUMPER WIRE CHANGE)
RAM PLATFORM DRIVE		x	TRW	2	FLTSATCOM	DELETE SLIP RINGS
STRUCTURE AND MECHANISMS						
STRUCTURE	x	x	TRW	3	NEW	EXISTING SPACECRAFT CONCEPTS
NUTATION DAMPER	x	x	TRW	3	NEW	SIMILAR TO METEOSAT DESIGN
MAGNETOMETER BOOM		x	CELESCO	3	NEW	BASED ON VIKING BOOM DESIGN
RELEASE MECHANISMS						
BALL LOCK MECHANISM	x			1	MINUTEMAN PROGRAM	NONE
PIN PULLERS	x		TRW	1	PIONEERS 10 AND 11	NONE

*1 = FLIGHT-PROVEN; 2 = DEVELOPED; 3 = NEW

6.2.2 Thor/Delta Configurations

T/D III T/D III 31 W T/D III 12 W T/D III T/D III

6.2.2.1 Mechanical Design Concept

The mechanical design concept for the spacecraft described in Section 6.2.1.1 for the Atlas/Centaur also applies to the Thor/Delta, except that the primary structure is defined by the accelerations at third-stage burnout.

An exploded view of the Thor/Delta probe bus spacecraft is shown in Figure 6-40. Figure 6-41 illustrates the simplicity and commonality of the structural design.

Mass properties of the spacecraft are tabulated in Section 6.2.2.3.

6.2.2.2 Dynamics and Attitude Control

The dynamics disturbances to the attitude control system for the Thor/Delta configuration are similar to those described in Section 6.2.1.2 for the Atlas/Centaur configuration.

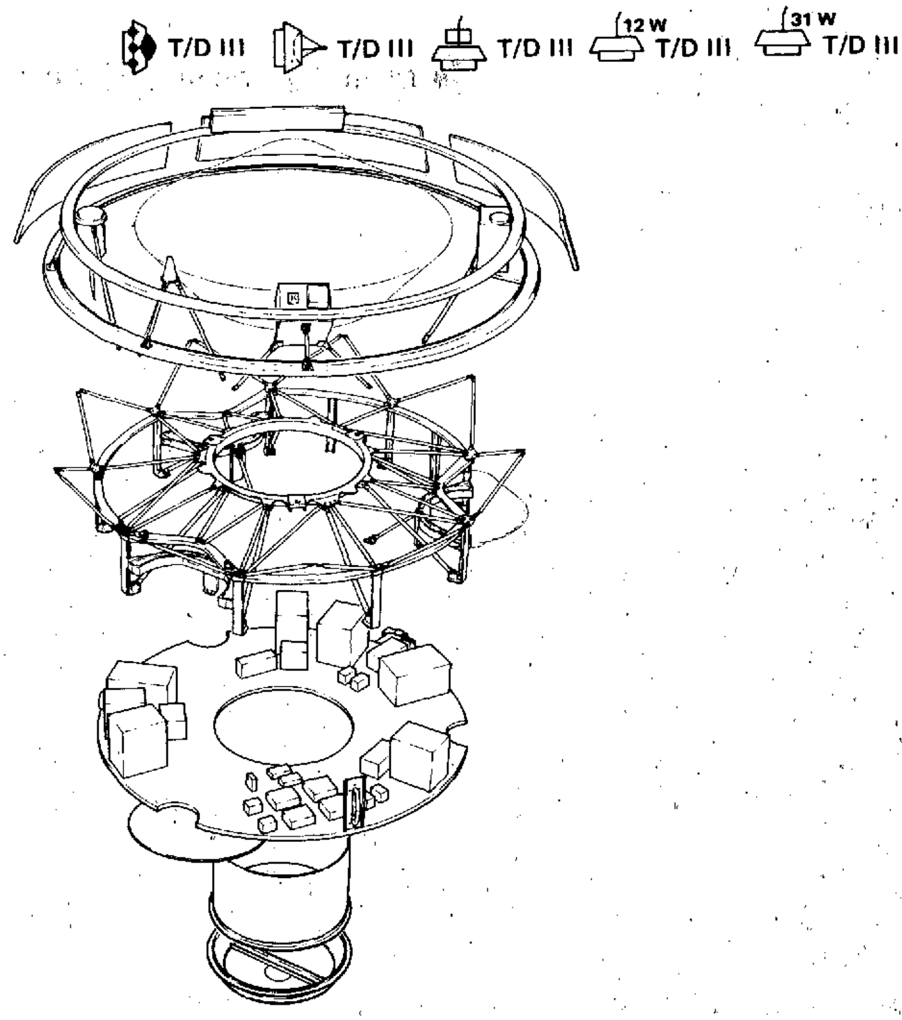


Figure 6-40. Exploded View of Thor/Delta Probe Bus Spacecraft

As discussed in Section 6.2.1.2, thrust-level differential produces a body-fixed transverse torque on the spacecraft that causes the spacecraft spin axis to cone in inertial space. Figures 6-42 and 6-43 illustrate this.

The dynamic disturbances for the Thor/Delta probe bus and orbiter are shown in Tables 6-20 and 6-21.

6.2.2.3 Mass Properties

Weight and mass properties estimates are summarized in this section for the preferred Thor/Delta probe and orbiter spacecraft configurations and the three optional orbiter configurations as described in Sections 5.2.3 and 5.2.4.

C-3

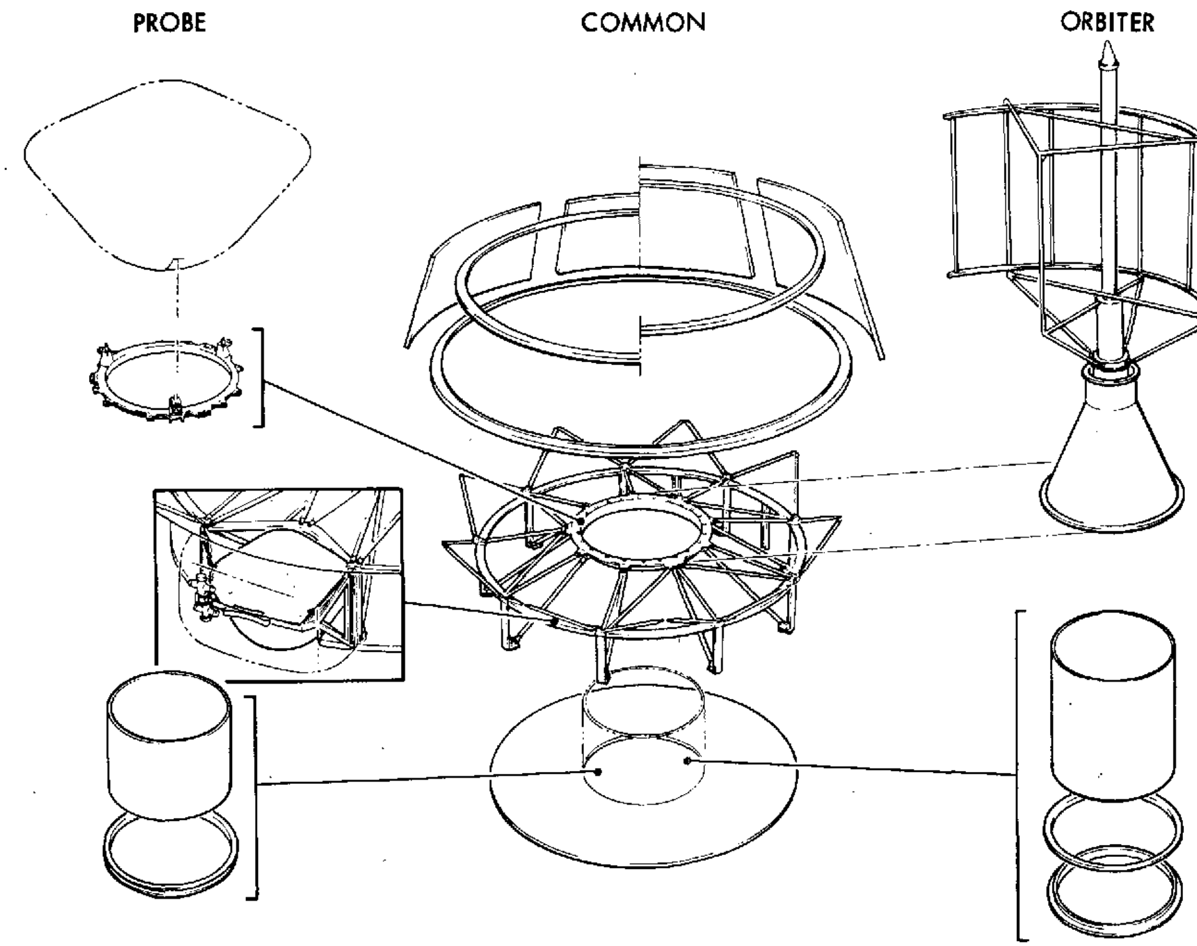


Figure 6-41. Simplicity and Commonality of Structural Design

T/D III
T/D III
T/D III
12W T/D III
31W T/D III

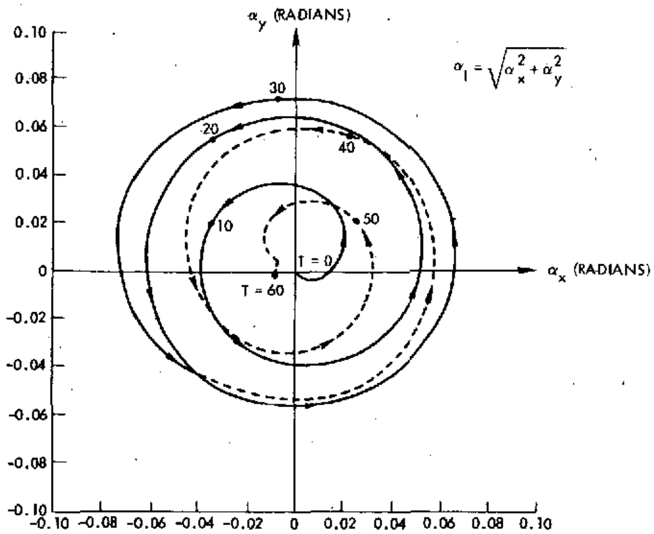


Figure 6-42. Example of Angle of Attack Trace in Inertial Space (Thor/Delta Probe, First Midcourse Maneuver)

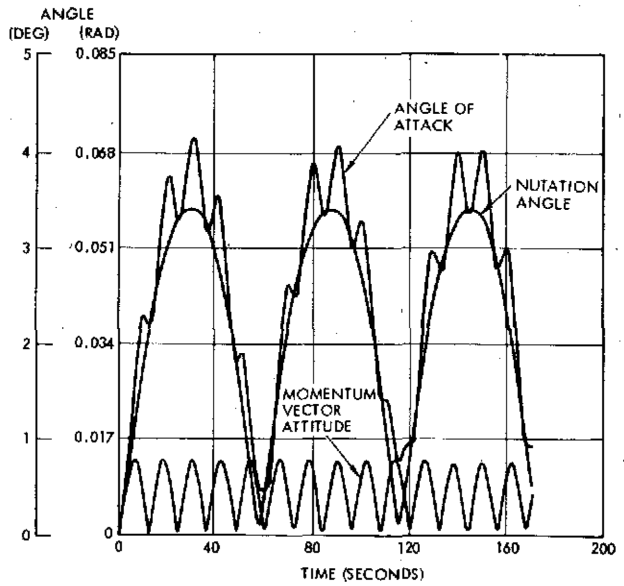


Figure 6-43. Examples of Amplitudes of Angle of Attack, Nutation Angle, and Momentum Vector Attitude Versus Time (Thor/Delta Probe, First Midcourse Maneuver)

Presented are mass properties requirements, weight summaries, detailed weight breakdowns, mass properties estimates for various flight conditions, and the coordinate reference axes and notation systems used in the mass properties analyses. Details of the large and small probe weights and mass properties used in this section are presented in Section 6.1.2.3.

Table 6-20. Thor/Delta Probe Dynamic Disturbances

T/D III

EVENT		ΔV (M/S)	THRUST EACH THRUSTER [N (LB)]	SPIN-RATE CHANGE [RAD/S (RPM)]	ANGLE-OF-ATTACK ERROR [RAD (DEG)]	MOMENTUM VECTOR SHIFT [RAD (DEG)]	VELOCITY DISPERSION ANGLE [RAD (DEG)]	VELOCITY DEGRADATION (M/S)	NUTATION ANGLE (DEG)
SEPARATION FROM BOOSTER		0.2		(0)		0.038 (2.2)			0.031 (1.8)
FIRST MIDCOURSE	(1)	73	5.2 (1.17)	±1.047 (±10)	0.077 (4.4)	0.014 (0.8)	0.007 (0.4)	0.1	0.065 (3.7)
SECOND MIDCOURSE	(1)	7	~3.1 (0.7)	±0.010 (±0.1)	0.035 (2.0)	0.009 (0.5)	0.004 (0.25)	0.004	0.038 (2.2)
THIRD MIDCOURSE	(1)	2	~3.1 (0.7)	±0.030 (±0.3)	0.045 (2.6)	0.009 (0.5)	0.004 (0.25)	0.001	0.038 (2.2)
	(2)	2	~3.1 (0.7)	±0.205 (±1.4)	2.443 (140)	2.443 (140)	LARGE		0.003 (0.2)
FIRST RETARGETING	(1)	1.02	~3.1 (0.7)	±0.010 (±0.1)	0.023 (1.3)	0.010 (0.6)	0.005 (0.3)	-	0.012 (0.7)
	(2)	1.02	~3.1 (0.7)	±0.042 (±0.4)	0.009 (0.5)	0.009 (0.5)	0.004 (0.25)		0.00007 (0.004)
SECOND RETARGETING	(1)	7.32	~3.1 (0.7)	±0.073 (±0.7)	0.209 (12)	0.070 (4)	0.035 (2)	0.041	0.140 (8)
THIRD RETARGETING	(1)	6.34	~3.1 (0.7)	±0.063 (±0.6)	0.140 (8)	0.087 (5)	0.045 (2.5)	0.066	0.070 (4)
	(2)	6.34	~3.1 (0.7)	±0.230 (±2.2)	0.075 (4.3)	0.075 (4.3)	0.045 (2.2)	-	NEGLIGIBLE
FOURTH RETARGETING	(1)	26.55	3.1 (0.7)	±0.293 (±2.8)	0.038 (2.2)	0.017 (1)	0.009 (0.5)	0.008	0.021 (1.2)

ASSUMES 0.503 RAD/S (4.8 RPM) SPIN RATE FOR ALL CASES

(1) USING PAIR OF AXIAL THRUSTERS WITH 9 MILLIRADIAN MISALIGNMENT

(2) USING PAIR OF SPIN THRUSTERS

Table 6-21. Thor/Delta Orbiter Dynamic Disturbances

T/D III

T/D III

12 W T/D III

31 W T/D III

EVENT	ΔV (M/S)	THRUST EACH THRUSTER [N (LB)]	SPIN-RATE CHANGE [RAD/S (RPM)]	ANGLE-OF-ATTACK ERROR [RAD (DEG)]	MOMENTUM VECTOR SHIFT [RAD (DEG)]	VELOCITY DISPERSION ANGLE [RAD (DEG)]	VELOCITY DEGRADATION (M/S)	NUTATION ANGLE (DEG)
SEPARATION FROM BOOSTER	0.25*		(0)		(5)	-		0.035 (2)
FIRST MIDCOURSE	73	~5.2 (1.17)	±1.309 (±12.5)	0.084 (4.8)	0.023 (1.3)	0.012 (0.7)	0.11	0.061 (3.5)
SECOND MIDCOURSE	7	~3.1 (~0.7)	±0.126 (±1.2)	0.051 (2.9)	0.014 (0.8)	0.007 (0.4)	0.004	0.037 (2.1)
THIRD MIDCOURSE	2	~3.1 (~0.7)	±0.037 (±0.35)	0.051 (2.9)	0.014 (0.8)	0.007 (0.4)	0.001	0.037 (2.1)
DEBOOST		28 500 (6400)	0 (0)	0.154 (8.8)	0.042 (2.4)	0.021 (1.2)	(0.5%)	0.112 (6.4)
PERIAPSIS TRIM ΔV (TOTAL)	43.5	~3.1 (~0.7)	±0.524 (±5)	0.037 (2.1)	0.010 (0.6)	0.005 (0.3)	0.013	0.026 (1.5)

*ASSUMES 1 M/S RELATIVE VELOCITY AT SEPARATION

OTHER ASSUMPTIONS: 9 MILLIRADIAN THRUSTER MISALIGNMENT
±4 PERCENT THRUST-LEVEL UNCERTAINTY FOR EACH THRUSTER
0.524 RAD/S (5 RPM) SPIN RATE DURING MIDCOURSE AND PERIAPSIS TRIM MANEUVERS
6.283 RAD/S (60 RPM) SPIN RATE DURING DEBOOST (VENUS ORBIT INSERTION)

Requirements

The probe and orbiter spacecraft mass properties requirements imposed by launch vehicle and mission considerations are summarized in Table 6-22.

Table 6-22. Thor/Delta Spacecraft Mass Properties Requirements, Version III Science Payload

PROPERTY	REQUIREMENT	
	PROBE SPACECRAFT	ORBITER SPACECRAFT
SPACECRAFT WEIGHT AFTER SEPARATION	385.1 KG (849.0 LB) MAXIMUM (TYPE I TRAJECTORY)	292.6 KG (645.0 LB) MAXIMUM (TYPE II TRAJECTORY
LONGITUDINAL CENTER-OF-GRAVITY LIMITATION (DISTANCE OF SPACECRAFT CENTER OF GRAVITY FORWARD OF THE SPACECRAFT/ THIRD-STAGE SEPARATION PLANE)	≤863.6 MM (34.0 IN.) BASED ON A 408.2 KG (900 LB) SPACECRAFT	
RADIAL CENTER-OF-GRAVITY OFFSET FROM SPACECRAFT CENTERLINE DURING LAUNCH CONDITIONS	≤0.381 MM (0.015 IN.), PER PHASE B RFP SPECIFICATIONS. NOTE:* PER DELTA RESTRAINTS DOCUMENT, THE STATIC BALANCE REQUIREMENT SPECIFIES ALLOWABLE OFFSET OF ≤1.27 MM (0.050 IN.)	
INERTIA RATIO (RATIO OF SPIN TO TRANSVERSE INERTIAS FOR LONG-DURATION SPIN-STABILITY CONSIDERATIONS)	>1.10	>1.10
SPACECRAFT PRINCIPAL SPIN AXIS PARALLEL TO SPACECRAFT LONGITUDINAL CENTERLINE DURING LAUNCH CONDITIONS	≤0.002 RADIAN, PER PHASE B RFP SPECIFICATIONS. NOTE:* PER DELTA DESIGN RESTRAINTS DOCUMENT, THE DYNAMIC BALANCE REQUIREMENT SPECIFIES ALLOWABLE PRINCIPAL SPIN-AXIS OFFSET OF 0.020 RADIAN	
PRINCIPAL SPIN AXIS IN THE XY AND XZ PLANE PARALLEL TO THE X-AXIS	≤0.0035 RADIAN (≤0.20 DEGREE) (PRELIMINARY ALLOCATION) DURING PROBE SEPARATION AND BUS REENTRY	≤0.0035 RADIAN (≤0.20 DEGREE) (PRELIMINARY ALLOCATION) DURING VENUS ORBIT OPERATIONS

*DAC-61687, "DELTA SPACECRAFT DESIGN RESTRAINTS," OCTOBER 1968, REVISED AUGUST 1972.

The static and dynamic balance constraints are based on the specification requirements stipulated in Appendix C of the Phase B RFP. Launch vehicle considerations impose requirements on the radial center-of-gravity offset and principal spin-axis orientation which are much tighter than the constraints currently specified by the launch vehicle supplier, as noted in the table.

Preferred 36-Watt Fanbeam, Fanscan Spacecraft Configuration

31 W
T/D III

The detailed weight breakdowns for the preferred probe and orbiter spacecraft configurations are presented in Figure 6-44A. The contingency margins, i.e., the net weight remaining for contingency provision) for the probe and orbiter are 24.8 kg (54.8 lb) and 10.0 kg (22.1 lb), respectively. As a percentage of dry spacecraft weight, this represents a contingency factor of 7.3 percent for the probe and 5.5 percent for the orbiter. Because preliminary contingency analyses, Appendix 6E, show an estimated 11.8 percent for the probe and 10.2 percent for the orbiter, both configurations present a weight problem. A weight reduction program could increase the contingency margin, but this sacrifices low cost considerations. Detailed weight/cost tradeoffs are discussed in Section 12.

Spacecraft mass properties characteristics for various flight conditions, summarized in Figure 6-44B, are based on coordinate reference axes and notation systems, Figure 6-44C. For long-duration spin-stability considerations, the inertia ratios (the ratio of spin to transverse inertias) are >1.10 for all conditions after booster separation. During third-stage TE364-4 burn conditions, the inertia ratio is <1.0 and is acceptable based on a short-duration spin-stability criteria.

The inertia parameters (λ) for the probe spacecraft before probe separation range in value between 0.15 and 0.19, and during and after probe separation between 0.61 and 0.77. A fine-tuned damper and a coarse-tuned damper satisfy the dynamic requirements during these two periods. For the orbiter spacecraft the inertia parameters range in value between 0.25 and 0.31. A single damper is used in this configuration.

To minimize principal spin-axis misalignment of the probe spacecraft during periods of asymmetric separation of the three small probes and the deployment of the science sensors before Venus entry, all expendables, deployables, and separable elements are located in the composite longitudinal center-of-gravity plane of the spacecraft bus (i.e., in the longitudinal center-of-gravity plane of the spacecraft less the large probe). Radial shift of the spin axis occurs during these periods; however, the spin axis remains parallel to the longitudinal reference axis since products of inertia are not induced in the XY and XZ planes.

A principal spin-axis misalignment condition exists, however, during transit conditions (before large probe separation) when the magnetometer sensor is deployed. Since the longitudinal center-of-gravity plane of the spacecraft (with large probe) is displaced from the bus center-of-gravity plane where the magnetometer is located, products of inertia are induced when the magnetometer is deployed. The spin-axis misalignment is 0.030 radian (~1.7 degrees) during this condition. Should this tilt in the spin axis be unacceptable during certain transit periods (e.g., during midcourse maneuvers), the magnetometer sensor can be retracted to the stowed condition to alleviate the misalignment. Figure 6-44B summarizes details relative to principal axis orientation.

A DETAILED WEIGHT SUMMARY

DESCRIPTION	WEIGHT: PROBE MISSION (KG)	(LB)	ORBITER MISSION (KG)	(LB)
ELECTRICAL POWER	15.7	34.7	42.1	92.8
SOLAR ARRAY ASSEMBLY (SIX PANELS)	6.80	15.0	15.88	35.0
BATTERY	1.59	3.5	17.42	38.4
POWER CONTROL UNIT INCLUDING SHUNTS	4.45	9.8	6.35	14.0
DC-DC CONVERTER	2.90	6.4	2.45	5.4
COMMUNICATIONS	9.4	20.8	15.9	35.0
CONSCAN PROCESSOR	-----	-----	0.36	0.8
RECEIVERS (2)	2.36	5.2	2.36	5.2
TRANSMITTER DRIVERS (2)	1.09	2.4	1.09	2.4
POWER AMPLIFIERS (2)	0.54	1.2	-----	-----
TWT	-----	-----	5.44	12.0
HYBRIDS (1/5)	0.05	0.1	0.23	0.5
DIPLEXERS (2)	1.95	4.3	1.95	4.3
SWITCHES (4/5)	1.09	2.4	1.36	3.0
FORWARD OMNI	0.14	0.3	0.14	0.3
AFT OMNI	0.41	0.9	0.23	0.5
MEDIUM GAIN ANTENNA	0.91	2.0	-----	-----
FAN-BEAM ANTENNA	-----	-----	1.13	2.5
FANSCAN ANTENNA	-----	-----	0.45	1.0
RF COAX AND CONNECTORS	0.91	2.0	1.13	2.5
ELECTRICAL DISTRIBUTION	12.2	26.8	12.6	27.8
COMMAND DISTRIBUTION UNIT	3.08	6.8	3.54	7.8
HARNESS AND CONNECTORS	9.07	20.0	9.07	20.0
DATA HANDLING	3.9	8.5	5.7	12.5
DIGITAL TELEMETRY UNIT	3.08	6.8	3.08	6.8
DIGITAL DECODER UNIT (2)	0.77	1.7	0.77	1.7
DATA STORAGE UNIT (3)	-----	-----	1.82	4.0
ATTITUDE CONTROL	2.3	5.1	2.3	5.1
CONTROL ELECTRONICS ASSEMBLY	1.91	4.2	1.91	4.2
SUN SENSOR ASSEMBLY (2)	0.41	0.9	0.41	0.9
PROPULSION (DRY)	6.8	14.9	6.8	14.9
PROPELLANT TANK ASSEMBLY (3)	3.13	6.9	3.13	6.9
THRUSTER ASSEMBLY (8)	2.18	4.8	2.18	4.8
FILTER	0.18	0.4	0.18	0.4
PRESSURE TRANSDUCER	0.18	0.4	0.18	0.4
FILL AND DRAIN VALVE ASSEMBLY	0.18	0.4	0.18	0.4
PROPELLANT LINES AND MISCELLANEOUS	0.91	2.0	0.91	2.0
SOLID INSERTION MOTOR (BURNOUT)	-----	-----	9.1	20.0
THERMAL CONTROL	10.4	23.0	14.9	32.9
INSULATION ASSEMBLY	5.81	12.8	5.44	12.0
FORWARD CLOSURE ASSEMBLY	0.86	1.9	1.00	2.2
SIDE CLOSURE ASSEMBLY	0.41	0.9	0.41	0.9
LOUVER ASSEMBLY (3/6 SQUARE FEET)	1.77	3.9	3.54	7.8
THERMAL FIN-TRANSMITTER	0.68	1.5	3.63	8.0
HEATERS, ISOLATORS, PAINT, ETC.	0.91	2.0	0.91	2.0
STRUCTURE	44.1	97.2	40.4	89.0
CENTRAL CYLINDER ASSEMBLY	9.71	21.4	10.25	22.6
UPPER RING	(2.68)	(5.9)	(2.09)	(4.6)
CYLINDER	(2.45)	(5.4)	(2.18)	(4.8)
PLATFORM SUPPORT RING	(4.31)	(9.5)	(0.59)	(1.3)
SEPARATION RING			(3.58)	(7.9)
MOTOR MOUNTING RING	-----	-----	(1.45)	(3.2)
ATTACH HARDWARE	(0.27)	(0.6)	(0.36)	(0.8)
STRUCTURE (CONTINUED)				
PLATFORM/COMPARTMENT ASSEMBLY	13.88	30.6	13.88	30.6
UPPER STRUTS (15)	(1.27)	(2.8)	(1.27)	(2.8)
PLATFORM STRUTS (9)	(0.86)	(1.9)	(0.86)	(1.9)
VERTICALS (9)	(1.04)	(2.3)	(1.04)	(2.3)
UPPER RING ASSEMBLY	(1.36)	(3.0)	(1.36)	(3.0)
PLATFORM STRUT FITTINGS (9)	(0.64)	(1.4)	(0.64)	(1.4)
PLATFORM ASSEMBLY	(7.76)	(17.1)	(7.76)	(17.1)
BRACKETS AND ATTACH HARDWARE	(0.95)	(2.1)	(0.95)	(2.1)
SOLAR ARRAY SUPPORT ASSEMBLY	3.76	8.3	3.49	7.7
UPPER RING	(1.27)	(2.8)	(1.00)	(2.2)
LOWER RING	(1.45)	(3.2)	(1.45)	(3.2)
STRUTS (18)	(1.04)	(2.3)	(1.04)	(2.3)
PROBE SUPPORT AND RELEASE MECHANISMS (4)	5.35	11.8	-----	-----
ANTENNA SUPPORT ASSEMBLY	-----	-----	3.18	7.0
MAGNETOMETER BOOM ASSEMBLY	2.72	6.0	2.72	6.0
PROPULSION SUPPORT ASSEMBLY	1.81	4.0	1.81	4.0
DAMPER (2/1)	1.81	4.0	1.81	4.0
FORWARD OMNI SUPPORT	0.23	0.5	-----	-----
AFT OMNI SUPPORT	0.23	0.5	0.23	0.5
MEDIUM GAIN ANTENNA SUPPORT	0.23	0.5	-----	-----
SCIENCE SUPPORT BRACKETRY	0.91	2.0	0.91	2.0
NM/IM SPECTROMETER SUPPORTS (2)	1.36	3.0	-----	-----
EQUIPMENT TIE-DOWN AND INTEGRATION HARDWARE	2.09	4.6	2.09	4.6
BALANCE WEIGHT PROVISION	2.7	6.0	2.7	6.0
• SPACECRAFT BUS LESS SCIENCE (DRY)	107.5	237.0	152.5	336.0
SCIENTIFIC INSTRUMENTS	11.2	24.6	28.3	62.5
NEUTRAL MASS SPECTROMETER	5.00	11.0	4.54	10.0
ION MASS SPECTROMETER	1.36	3.0	1.36	3.0
ELECTRON TEMPERATURE PROBE	1.13	2.5	1.13	2.5
UV FLUORESCENCE	1.36	3.0	-----	-----
MAGNETOMETER	2.31	5.1	2.27	5.0
UV SPECTROMETER	-----	-----	5.44	12.0
IR RADIOMETER	-----	-----	4.08	9.0
RF ALTIMETER	-----	-----	9.53	21.0
• SPACECRAFT BUS (DRY)	118.7	261.6	180.8	398.5
PROBES (WITHOUT CONTINGENCY)	222.2	489.8	-----	-----
LARGE PROBE	143.2	315.7	-----	-----
SMALL PROBES (3)	79.0	174.1	-----	-----
• SPACECRAFT (DRY)	340.9	751.4	180.8	398.5
PROPELLANTS AND PRESSURANT	19.4	42.8	101.8	224.4
INSERTION PROPELLANT AND EXPENDED INERTS	-----	-----	84.41	186.1
HYDRAZINE PROPELLANT	19.14	42.2	17.10	37.7
NITROGEN PRESSURANT	0.27	0.6	0.27	0.6
• SPACECRAFT LESS CONTINGENCY	360.3	794.2	282.6	622.9
CONTINGENCY (NET ALLOWABLE)	24.8	54.8	10.0	22.1
(PERCENT OF DRY SPACECRAFT WEIGHT)	(7.3)		(5.5)	
• GROSS SPACECRAFT AFTER SEPARATION	385.1	849.0	292.6	645.0

B SUMMARY OF SPACECRAFT MASS PROPERTIES CHARACTERISTICS DURING VARIOUS FLIGHT CONDITIONS

CONDITION	WEIGHT (KG)	WEIGHT (LB)	CENTER OF GRAVITY X (CM)	X (IN)	Y (CM)	Y (IN)	Z (CM)	Z (IN)
PROBE MISSION								
AT TE-364-4 IGNITION	1527.2	3367	201.7	79.4	0	0	0	0
AT TE-364-4 BURNOUT	484.0	1067	276.9	109.0	0	0	0	0
AFTER THIRD STAGE SEPARATION (MAGNETOMETER STOWED)	385.1	849	308.6	121.5	0	0	0	0
AFTER THIRD STAGE SEPARATION (MAGNETOMETER DEPLOYED)	385.1	849	308.6	121.5	0.86	0.34	0	0
AFTER MIDCOURSE (MAGNETOMETER DEPLOYED)	370.1	816	309.6	121.9	0.89	0.35	0	0
PRIOR PROBE SEPARATION (MAGNETOMETER STOWED)	370.1	816	309.6	121.9	0	0	0	0
AFTER LARGE PROBE SEPARATION (MAGNETOMETER STOWED)	215.5	475	281.9	111.0	0	0	0	0
AFTER FIRST SMALL PROBE SEPARATION (MAGNETOMETER STOWED)	186.0	410	281.9	111.0	11.18	4.40	6.43	2.53
AFTER SECOND SMALL PROBE SEP. (MAGNETOMETER STOWED)	156.9	346	281.9	111.0	0	0	15.24	6.00
AFTER THIRD SMALL PROBE SEPARATION (MAGNETOMETER STOWED)	126.6	279	281.9	111.0	0	0	0	0
END OF LIFE (MAGNETOMETER DEPLOYED)	126.6	279	281.9	111.0	2.62	1.03	0	0
ORBITER MISSION								
AT TE-364-4 IGNITION	1434.7	3163	194.1	76.4	0	0	0	0
AT TE-364-4 BURNOUT	391.4	863	266.4	104.9	0	0	0	0
AFTER THIRD STAGE SEPARATION (MAGNETOMETER STOWED)	292.6	645	304.5	119.9	0	0	0	0
AFTER THIRD STAGE SEPARATION (MAGNETOMETER DEPLOYED)	292.6	645	304.5	119.9	1.14	0.45	0	0
AFTER MIDCOURSE (MAGNETOMETER DEPLOYED)	280.8	619	304.5	119.9	1.17	0.46	0	0
PRIOR VENUS ORBIT INSERTION (MAGNETOMETER STOWED)	280.8	619	304.5	119.9	0	0	0	0
AFTER VENUS ORBIT INSERTION (MAGNETOMETER STOWED)	196.4	433	304.5	119.9	0	0	0	0
AFTER VENUS ORBIT INSERTION (MAGNETOMETER DEPLOYED)	196.4	433	304.5	119.9	1.68	0.66	0	0
END OF LIFE (MAGNETOMETER DEPLOYED)	191.4	422	304.5	119.9	1.73	0.68	0	0

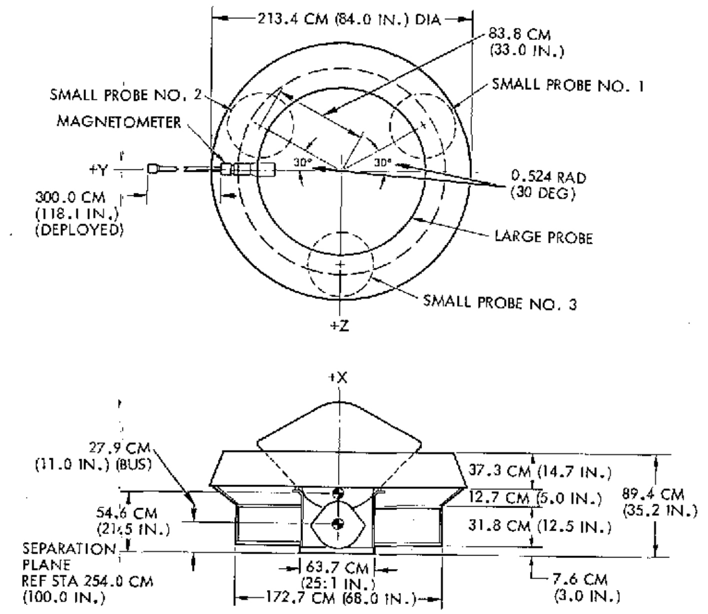
CONDITION	MOMENTS OF INERTIA(1) I_X (SPIN) (KG-M²)	I_X (SLUG FT²)	I_Y (KG-M²)	I_Y (SLUG FT²)	I_Z (KG-M²)	I_Z (SLUG FT²)	PRODUCTS OF INERTIA(1) P_{XY} (KG-M²)	P_{XY} (SLUG FT²)	P_{XZ} (KG-M²)	P_{XZ} (SLUG FT²)	P_{YZ} (KG-M²)	P_{YZ} (SLUG FT²)	RATIO(1) I_X/I_Y	I_X/I_Z	(1)(2) λ, INERTIA PARAMETER
PROBE MISSION															
AT TE-364-4 IGNITION	259.8	191.6	887.2	654.4	887.2	654.4	--	--	--	--	--	--	0.29	0.29	--
AT TE-364-4 BURNOUT	155.8	114.9	353.6	260.8	353.6	260.8	--	--	--	--	--	--	0.44	0.44	--
AFTER THIRD STAGE SEPARATION (MAGNETOMETER STOWED)	144.9 (144.9)	106.9 (106.9)	125.8 (122.2)	92.8 (90.1)	126.0 (129.8)	92.9 (95.7)	0	0	0	0	-3.81	-2.81 (-44.7°)	(1.19)	(1.12)	(0.15)
AFTER THIRD STAGE SEPARATION (MAGNETOMETER DEPLOYED)	158.8 (158.8)	117.1 (117.1)	125.8 (124.9)	92.8 (92.1)	139.8 (140.7)	103.1 (103.8)	-0.88 (+1.6°)	-0.65	0	0	-3.81	-2.81 (-14.5°)	(1.27)	(1.13)	(0.19)
AFTER MIDCOURSE (MAGNETOMETER DEPLOYED)	155.4 (155.4)	114.6 (114.6)	123.1 (122.0)	90.8 (90.0)	136.9 (137.9)	101.0 (101.7)	-0.92 (+1.7°)	-0.68	0	0	-3.81	-2.81 (-14.5°)	(1.27)	(1.13)	(0.19)
PRIOR PROBE SEPARATION (MAGNETOMETER STOWED)	141.5 (141.5)	104.4 (104.4)	123.1 (119.3)	90.8 (88.0)	123.1 (126.9)	90.8 (93.6)	0	0	0	0	-3.81	-2.81 (-44.7°)	(1.19)	(1.12)	(0.15)
AFTER LARGE PROBE SEPARATION (MAGNETOMETER STOWED)	125.3 (125.3)	92.4 (92.4)	71.2 (74.4)	52.5 (54.9)	70.1 (66.8)	51.7 (49.3)	0	0	0	0	-3.81	-2.81 (40.8°)	(1.68)	(1.87)	(0.77)
AFTER FIRST SMALL PROBE SEPARATION (MAGNETOMETER STOWED)	101.4 (101.4)	74.8 (74.8)	64.9 (73.8)	47.9 (54.4)	52.2 (43.4)	38.5 (32.0)	0	0	0	0	-13.84	-10.21 (32.7°)	(1.38)	(2.34)	(0.71)
AFTER SECOND SMALL PROBE SEP. (MAGNETOMETER STOWED)	80.1 (80.1)	59.1 (59.1)	56.5 (57.4)	41.7 (42.3)	39.0 (38.2)	28.8 (28.2)	0	0	0	0	-3.81	-2.81 (11.7°)	(1.40)	(2.10)	(0.66)
AFTER THIRD SMALL PROBE SEPARATION (MAGNETOMETER STOWED)	62.9 (62.9)	46.4 (46.4)	39.6 (42.8)	29.2 (31.6)	38.5 (35.1)	28.4 (25.9)	0	0	0	0	-3.81	-2.81 (40.7°)	(1.47)	(1.79)	(0.61)
END OF LIFE (MAGNETOMETER DEPLOYED)	76.6 (76.6)	56.5 (56.5)	39.6 (38.5)	29.2 (28.4)	52.2 (53.3)	38.5 (39.3)	0	0	0	0	-3.81	-2.81 (-15.6°)	(1.99)	(1.44)	(0.66)
ORBITER MISSION															
AT TE-364-4 IGNITION	204.0	150.5	690.1	509.0	695.0	512.6	--	--	--	--	--	--	0.30	0.29	--
AT TE-364-4 BURNOUT	100.1	73.8	275.1	202.9	280.0	206.5	--	--	--	--	--	--	0.36	0.36	--
AFTER THIRD STAGE SEPARATION (MAGNETOMETER STOWED)	89.3 (89.3)	65.9 (65.9)	68.1 (67.5)	50.2 (49.8)	72.9 (73.5)	53.8 (54.2)	0	0	0	0	-1.75	-1.29 (-17.7°)	(1.32)	(1.22)	(0.26)
AFTER THIRD STAGE SEPARATION (MAGNETOMETER DEPLOYED)	103.0 (103.0)	76.0 (76.0)	68.1 (67.9)	50.2 (50.1)	86.6 (86.9)	63.9 (64.1)	0	0	0	0	-1.75	-1.29 (-5.3°)	(1.52)	(1.19)	(0.31)
AFTER MIDCOURSE (MAGNETOMETER DEPLOYED)	100.3 (100.3)	74.0 (74.0)	66.7 (66.6)	49.2 (49.1)	85.3 (85.6)	62.9 (63.1)	0	0	0	0	-1.75	-1.29 (-5.3°)	(1.51)	(1.17)	(0.30)
PRIOR VENUS ORBIT INSERTION (MAGNETOMETER STOWED)	86.6 (86.6)	63.9 (63.9)	66.7 (66.2)	49.2 (48.8)	71.6 (72.1)	52.8 (53.2)	0	0	0	0	-1.75	-1.29 (-17.7°)	(1.31)	(1.20)	(0.25)
AFTER VENUS ORBIT INSERTION (MAGNETOMETER STOWED)	84.5 (84.5)	62.3 (62.3)	64.4 (63.9)	47.5 (47.1)	69.3 (70.0)	51.1 (51.6)	0	0	0	0	-1.75	-1.29 (-17.7°)	(1.32)	(1.21)	(0.26)
AFTER VENUS ORBIT INSERTION (MAGNETOMETER DEPLOYED)	98.3 (98.3)	72.5 (72.5)	64.4 (64.3)	47.5 (47.4)	83.1 (83.2)	61.3 (61.4)	0	0	0	0	-1.75	-1.29 (-5.3°)	(1.53)	(1.18)	(0.31)
END OF LIFE (MAGNETOMETER DEPLOYED)	97.2 (97.2)	71.7 (71.7)	63.9 (63.7)	47.1 (47.0)	82.6 (82.7)	60.9 (61.0)	0	0	0	0	-1.75	-1.29 (-5.3°)	(1.53)	(1.18)	(0.30)

NOTE: (1) VALUES IN PARENTHESES ARE REFERENCED RELATIVE TO THE PRINCIPAL AXES.

(2) INERTIA PARAMETER, $\lambda = \left[\left(\frac{I_X - I_Y}{I_Y}\right)\left(\frac{I_X - I_Z}{I_Z}\right)\right]^{0.5}$

C COORDINATE REFERENCE AXES AND NOTATION SYSTEM

PROBE SPACECRAFT



ORBITER SPACECRAFT

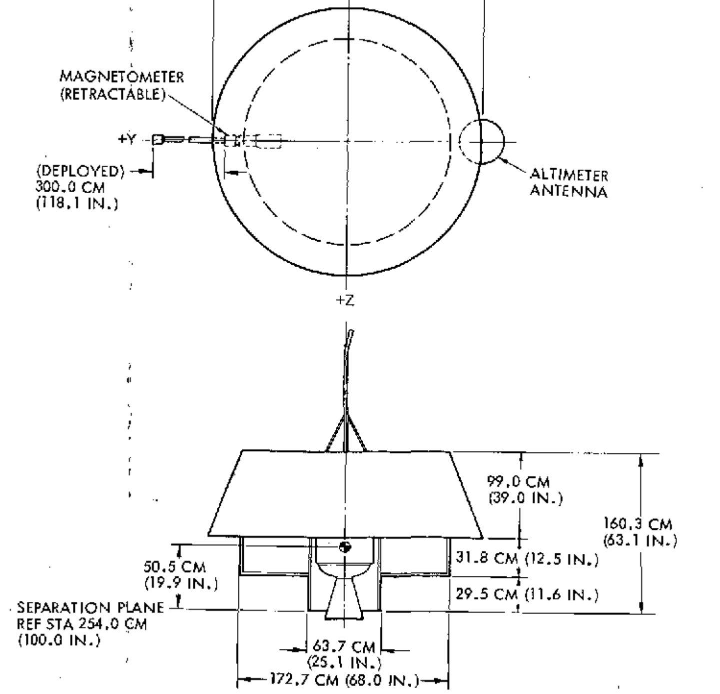


Figure 6-44. Preferred Fanbeam, Fanscan Thor/Delta Spacecraft Mass Properties Summary for Version III Science Payload and 1977/1978 Mission Launches

T/D III The orbiter spacecraft, a simpler configuration with no separable elements, is complicated by the current deployment method of the altimeter science antenna (out-of-plane deployment). To meet TE364-4 third-stage mass properties requirements and operational spacecraft requirements, the antenna, when deployed, balances the spacecraft statically and dynamically. The antenna is deployed after second-stage separation and before third-stage burn. An alternate method would deploy the antenna in the center-of-gravity plane, eliminating products of inertia problems in the XY and XZ planes. However, this would require better definition of the antenna assembly size and shape for stowing considerations between the spacecraft compartment and the shroud.

31 W T/D III

For both the probe and orbiter spacecraft configurations, the launch vehicle mass properties requirements were the most stringent. The probe configuration posed the most difficult problem; five separate entities (the four probes and the bus) had to be combined to meet not only the launch requirement but also the spacecraft requirements during sequential separation of the probes. A preliminary analysis, Appendix 6G, determined the allowable probe and bus mass properties uncertainties necessary for launch vehicle performance. Two sets of launch vehicle requirements were addressed: requirements per the Phase B RFP and per the launch vehicle supplier.

<u>Optional Orbiter Configurations</u> T/D III T/D III 12 W T/D III

A weight summary comparing the three optional orbiter configurations relative to the preferred configuration is presented in Table 6-23. The net allowable contingency for each of the optional configurations is greater than that for the preferred. Only Option 1 (12-watt fanbeam, fanscan) and Option 2 (earth-pointing) configurations provide contingency margins in excess of the 10.2 percent contingency factor which is assumed required per the preliminary contingency analyses (Appendix 6E).

All the optional configurations are spin-stable and have the same mass properties control considerations as the preferred configuration. The detailed mass properties for the optional Thor Delta configurations are summarized in Appendix 6H.

Table 6-23. Thor/Delta Optional Orbiter Configuration Weight Comparison Summary

DESCRIPTION	31 W T/D III		12 W T/D III		T/D III		T/D III	
	KG	(LB)	KG	(LB)	KG	(LB)	KG	(LB)
ELECTRICAL POWER	42.1	(92.8)	30.4	(66.9)	30.4	(66.9)	34.7	(76.4)
COMMUNICATIONS	15.9	(35.0)	11.5	(25.4)	15.4	(34.0)	12.4	(27.4)
ELECTRICAL DISTRIBUTION	12.6	(27.8)	12.6	(27.8)	12.6	(27.8)	12.6	(27.8)
DATA HANDLING	5.7	(12.5)	5.7	(12.5)	5.7	(12.5)	5.7	(12.5)
ATTITUDE CONTROL	2.3	(5.1)	2.3	(5.1)	2.3	(5.1)	12.4	(27.4)
PROPULSION (DRY)	6.8	(14.9)	6.8	(14.9)	6.8	(14.9)	6.8	(14.9)
SOLID INSERTION MOTOR (BURNOUT)	9.1	(20.0)	9.1	(20.0)	9.1	(20.0)	9.1	(20.0)
THERMAL CONTROL	14.9	(32.9)	11.2	(24.8)	11.2	(24.8)	11.2	(24.8)
STRUCTURE	40.4	(89.0)	41.9	(92.3)	44.2	(97.4)	40.9	(90.1)
BALANCE WEIGHT PROVISION	2.7	(6.0)	2.7	(6.0)	2.7	(6.0)	2.7	(6.0)
SPACECRAFT BUS LESS SCIENCE (DRY)	152.5	(336.0)	134.2	(295.7)	140.4	(309.4)	148.5	(327.3)
SCIENTIFIC INSTRUMENTS	28.3	(62.5)	28.3	(62.5)	28.3	(62.5)	28.3	(62.5)
SPACECRAFT (DRY)	180.8	(398.5)	162.5	(358.2)	168.7	(371.9)	176.8	(389.8)
INSERTION MOTOR EXPENDABLES	84.4	(186.1)	84.4	(186.1)	84.4	(186.1)	84.4	(186.1)
HYDRAZINE PROPELLANT AND PRESSURANT	17.4	(38.3)	17.4	(38.3)	17.4	(38.3)	17.4	(38.3)
SPACECRAFT LESS CONTINGENCY	282.6	(622.9)	264.3	(582.6)	270.5	(596.3)	278.6	(614.2)
CONTINGENCY (NET ALLOWABLE)	10.0	(22.1)	28.3	(62.4)	22.1	(48.7)	14.0	(30.8)
(PERCENT OF DRY SPACECRAFT WEIGHT)	(5.5%)		(17.4%)		(13.1%)		(7.9%)	
GROSS SPACECRAFT AFTER SEPARATION	292.6	(645.0)	292.6	(645.0)	292.6	(645.0)	292.6	(645.0)

6.2.2.4 Electrical Design Concept T/D III 31 W T/D III

The preferred Thor/Delta configuration is based on Version III scientific instrument requirements. The functional relationships and subsystem interconnections are essentially the same as for the Atlas/Centaur configuration (earth-pointing) described in Section 6.2.1.4, except for the selection of antennas for an orbiter spacecraft which has the spin axis oriented normal to the ecliptic plane. Moreover, the stringent weight and volume constraints imposed by the limited payload capability of this launch vehicle necessitated consideration of lighter, but possibly more costly, designs.

Figure 6-45 shows a functional block diagram for the probe bus. A comparable drawing is given in Figure 6-46 for the orbiter. The major deviations in electrical hardware implementation and/or design approach for the Thor/Delta configuration are summarized in the following paragraphs; detailed discussions of the tradeoffs and analyses leading to these recommendations are provided in Section 8.

Electrical Power

The inverter-CTRF assembly recommended for the Atlas/Centaur version is replaced with parallel redundant DC-DC equipment converters

in a single package. Figure 6-47 illustrates the concept. This selection is dictated by the weight savings (5. 7 kg) which can be achieved by combining transformers into one multiple secondary power transformer per channel. The dissipative-type output regulators provide excellent dynamic response to load variations as well as current limiting, in the event of overload, to protect the converter.

The battery designs are similar to those proposed for the Atlas/Centaur except with reduced capacity because of the smaller load requirements.

Communications

The more modest telemetry transmission rate requirements dictated by the Version III science payload led to an orbiter spacecraft configuration with the spin axis oriented perpendicular to the ecliptic plane, permitting adequate downlink coverage with a Franklin-array antenna. Derived directly from the Pioneers 6 through 9 Program, it has a fanbeam pattern 0. 096 radian (5. 5-degree) beamwidth with directionality normal to the spin axis. Coupled with a 31-watt (minimum) power amplifier, it provides sufficient downlink EIRP to support 8 bits/s at maximum range when a 26-meter deep space station is used for data acquisition. This antenna is coupled to either power amplifier via a transfer switch, thus avoiding the losses associated with a diplexer or hybrid.

A shorter version of this Franklin array, incorporating half the number of dipole elements, is tilted approximately 0. 061 radian (3. 5 degrees) with respect to the spin axis to generate a fanscan signal analogous to the conscan signal for Pioneers 10 and 11. In this case, however, the closed-loop attitude control system seeks to orient the spin axis perpendicular to the earth-spacecraft line. This antenna, attached to the top of the main downlink antenna, is interconnected to provide uplink reception only.

Communication coverage during the launch phase, midcourse maneuvers, and orbit insertion is supported by a Pioneer 6 through 9 slot array omni antenna (toroidal pattern with the axis of revolution coincident with the spin axis), and an aft-facing low-gain antenna. The

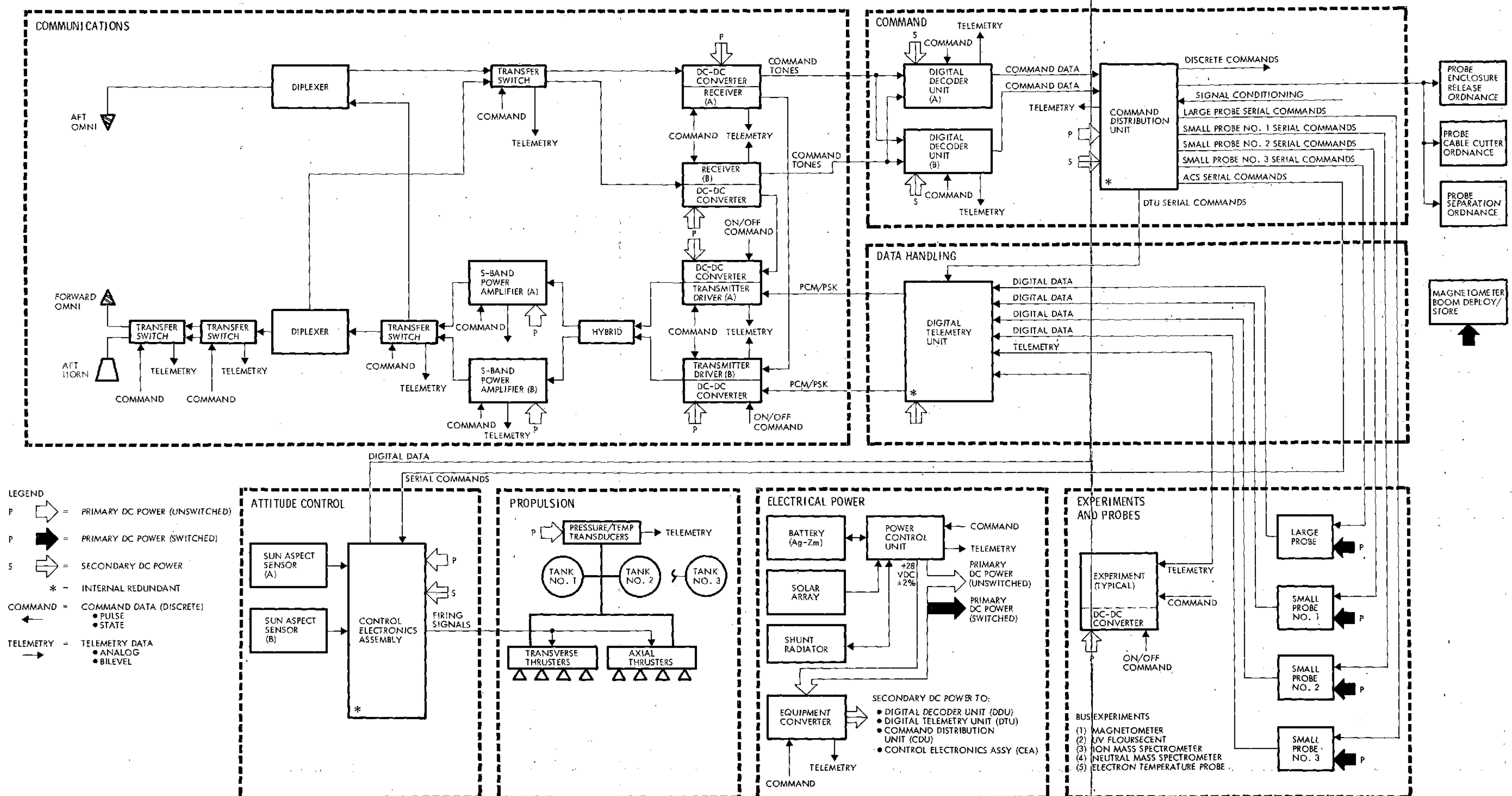


Figure 6-45. Probe Bus Functional Block Diagram, Preferred Thor/Delta Configuration

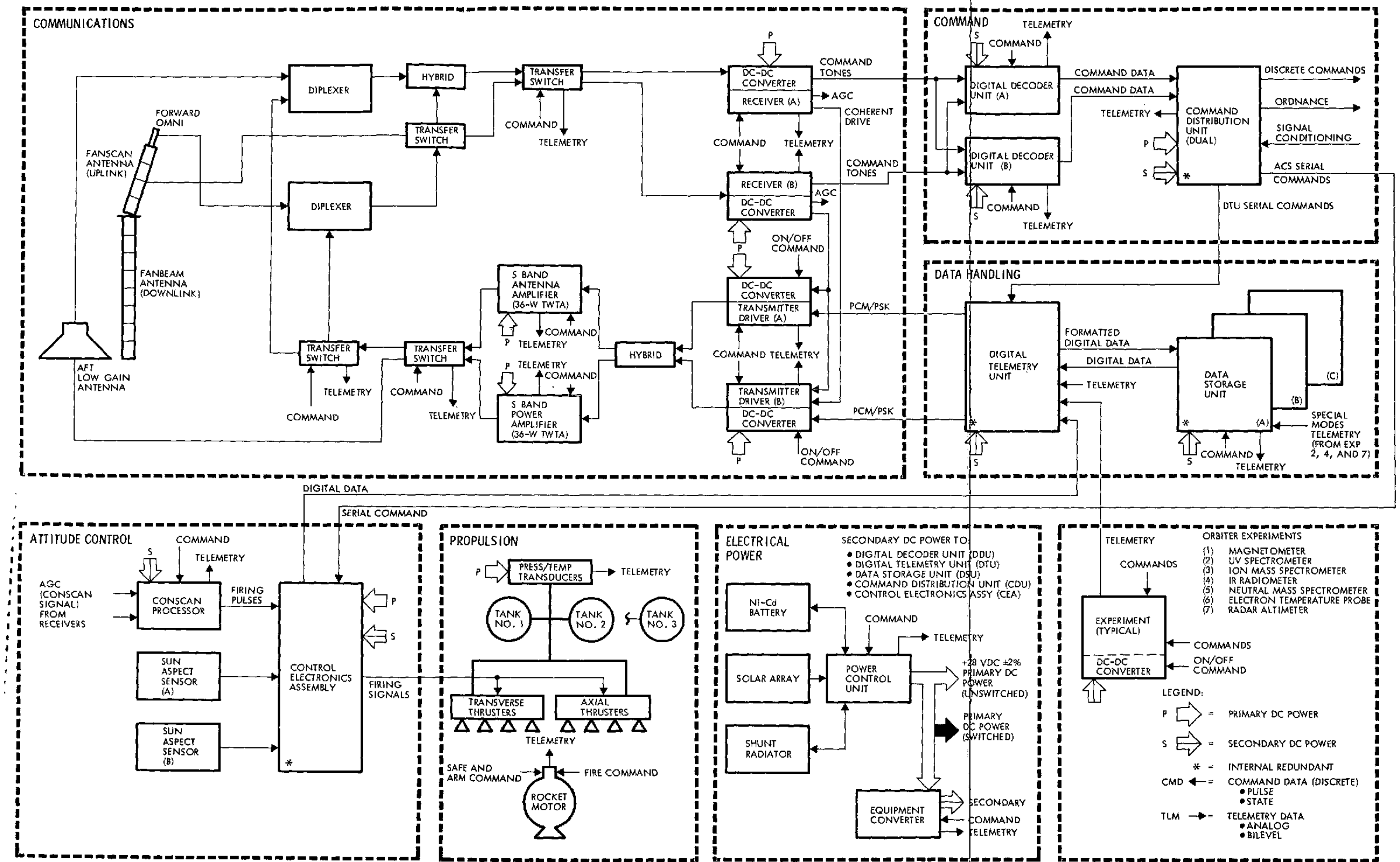


Figure 6-46. Orbiter Functional Block Diagram, Preferred Thor/Delta Configuration

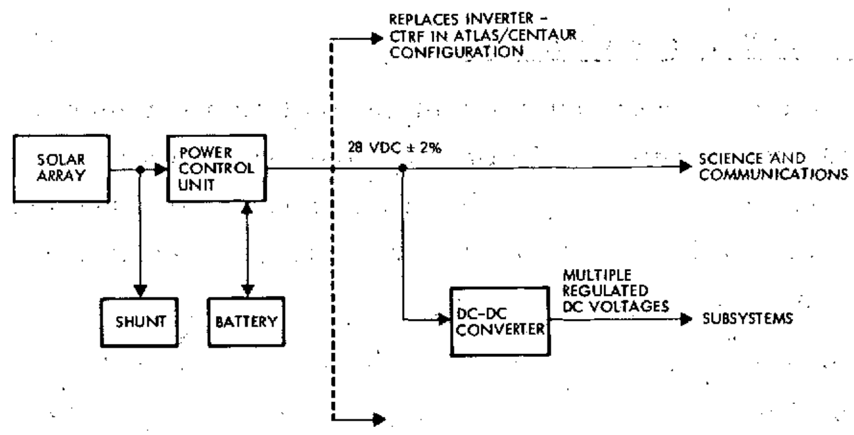


Figure 6-47. Thor/Delta Power Subsystem

Pioneer 10 and 11 high gain antenna feed has been selected to fulfill the latter function. Both antennas are configured to permit both uplink reception and downlink transmission. Figure 6-48 illustrates the recommended coupling of the four antennas with the redundant receivers and transmitters.

Stringent weight constraints for the Thor/Delta configuration necessitated restricting our consideration to micro-miniature transponders, where significant weight savings can be realized. Therefore, the use of residual Pioneer 10 and 11 receivers and drivers, while cost-effective for the Atlas/Centaur probe bus, is not recommended because of the excessive weight (2.7 kg differential). With this exception, the preferred configuration for the probe bus is identical to that given for the Atlas/Centaur.

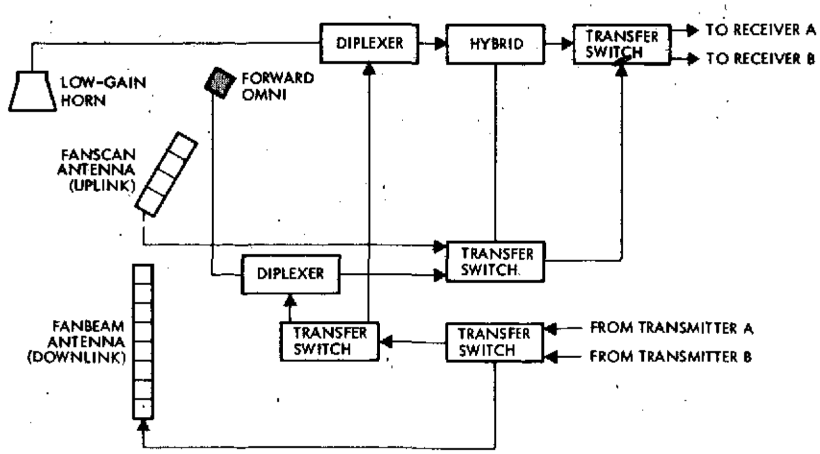
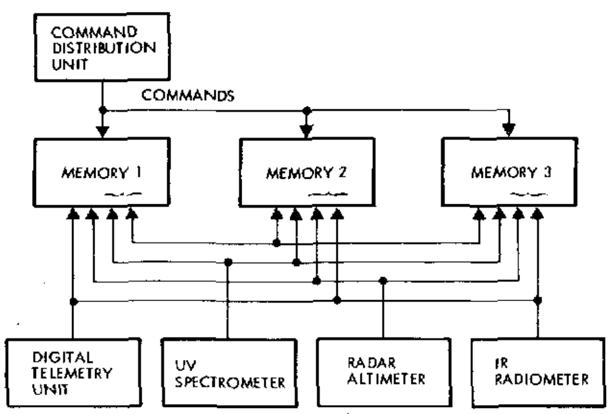


Figure 6-48. Functional Arrangement of Antennas, Preferred Thor/Delta Configuration

Data Handling and Command

The functional relationship and general performance of the data handling subsystem remain similar to those proposed for the Atlas/Centaur configuration. However, the Version III science requirements closely match the Pioneer 10 and 11 DTU capabilities, necessitating only the inclusion of an 8 bit/s data rate.

Whereas five DSU's were required to meet the data storage requirements of the Version IV science payload on the Atlas/Centaur configuration, three units provide the Thor/Delta with adequate redundancy and multiple simultaneous buffers to the scientific instruments. Figure 6-49 illustrates the reconfigurable redundancy approach selected.



COMMAND \ MEMORY	1	2	3
A	DTU	UV	RADAR/IR
B	RADAR/IR	DTU	UV
C	UV	RADAR/IR	DTU

Figure 6-49. Data Storage Concept, Preferred Thor/Delta Configuration

Each DSU contains two modules of 122,880 bits each. These modules can be chained together to form a single bank of 245,760 bits or they can be used independently. In this manner, all four data sources can be accommodated simultaneously. If a DSU fails, the instrument memory assignments are reconfigured, permitting any two of the three functions, as shown in the matrix. Redundancy is thus provided for the occultation period information and the radar altimeter data, both in-line functions. The UV spectrometer and IR radiometer experiment data return are enhanced by the availability of the added storage.

The Pioneer 10 and 11 ordnance firing circuits, designed to operate with 30.5 VAC (rms), must be redesigned to accept 28 VDC. Substitution of a DC-DC equipment converter for the inverter-CTRF necessitated this change.

Propulsion

The reaction control system (RCS) is configured identical to that proposed for the Atlas/Centaur version. However, Thor/Delta missions involve increased propellant allocation, primarily because of larger midcourse maneuvers required to correct the greater injection error. The recommended tanks from the DSCS-II Program can accommodate the increased hydrazine load.

The Hercules BE-3-A solid rocket capabilities closely approximate the Thor/Delta orbiter insertion impulse requirements. This motor has a demonstrated history of flight reliability with the Vela, Ranger, Athena, and Sparta programs. The existing design has no critical design limitations which would be exceeded by Pioneer Venus requirements. Moreover, only 4.1 percent propellant off-loading is required.

Electrical Distribution and Grounding

Figures 6-45 and 6-46 are the electrical distribution diagrams for the probe bus and orbiter, respectively. The implementation scheme is identical to the Atlas/Centaur (reference Figures 6-30 and 6-31) except for 1) replacement of the inverter-CTRF with a DC-DC equipment converter, and 2) substitution of the Viking transponder, which includes its own converter, for the Pioneer 10 and 11 receiver driver, which is dependent on conditioned secondary power.

Equipment Derivation/Status

Table 6-24 summarizes the selected hardware units for the probe bus and orbiter spacecraft. The subsystem components are identified by the categories of 1) flight-proven, developed, or 2) new. Their previous program application is also given.

Comparison of this list with that compiled for the Atlas/Centaur configuration (Table 6-18) reveals few changes with the exception of the orbiter antenna complement which relies on Pioneer 6 through 9 technology. The majority of the recommended equipment is flight-proven on previous Pioneer programs.

Table 6-24. Equipment Derivation and Status, Preferred Thor/Delta Configuration

SUBSYSTEM/COMPONENT	APPLICATION: PROBE BUS	APPLICATION: ORBITER	MANUFACTURER	STATUS*	PREVIOUS USAGE	MODIFICATIONS REQUIRED
COMMUNICATIONS						
TRANSPONDER (RECEIVER-DRIVER)	x	x	PHILCO-FORD	2	VIKING, SKYNET	INCLUDE CONSCAN AGC; MINOR INTERFACE MODIFICATIONS
POWER AMPLIFIER (SOLID STATE)	x		MICROWAVE SEMICONDUCTOR CORP.	2	COMMERCIAL APPLICATIONS	POSSIBLE GAIN MODIFICATION; INCLUSION OF HIGH-RELIABILITY AND SPACE QUALIFICATION REQUIREMENTS
POWER AMPLIFIER (31-W TWTA)		x	WATKINS-JOHNSON	1	PIONEERS 10 AND 11	NONE
DIPLEXER	x	x	WAVECOM	1	PIONEERS 10 AND 11	NONE
TRANSFER SWITCH	x	x	TELEDYNE	1	PIONEERS 10 AND 11	NONE
HYBRID COUPLER	x	x	ANAREN	2	NONE	SCALE FROM EXISTING L-BAND TO S-BAND
FRANKLIN ARRAY (20 ELEMENT)		x	TRW	1	PIONEERS 6 THROUGH 9	NONE
FRANKLIN ARRAY (10 ELEMENT)		x	TRW	2	PIONEERS 6 THROUGH 9	DELETE 10 ELEMENTS FROM PIONEERS 6 THROUGH 9 DESIGN
S-BAND MEDIUM-GAIN HORN	x	x	TRW	1	PIONEERS 10 AND 11	NONE
FORWARD OMNI	x		TRW	1	PIONEERS 10 AND 11	NONE
AFT OMNI	x		TRW	1	DEFENSE SUPPORT PROGRAM	NOT APPLICABLE
DATA HANDLING						
DIGITAL TELEMETRY UNIT	x	x	TRW	1	PIONEERS 10 AND 11	REDUCE LOWEST BIT RATE FROM 16 TO 8 BITS/S; PROVIDE GROUND-COMMANDED TWO-LEVEL MODULATION INDEX CONTROL
DATA STORAGE UNIT		x	TRW	3	NONE	NEW DEVELOPMENT USING EXISTING TECHNOLOGY
COMMAND						
DIGITAL DECODER UNIT	x	x	TRW	1	PIONEERS 10 AND 11	NONE
COMMAND DISTRIBUTION UNIT	x	x	TRW	1	PIONEERS 10 AND 11	AUGMENT ORDNANCE FIRING CIRCUITS; INCREASE COMMAND STORAGE CAPABILITY; ADD ORBIT INSERTION MOTOR FIRING LOGIC; REVISE ORDNANCE FIRING CIRCUITRY TO USE DC VOLTAGE
THERMAL						
LOUVERS	x	x	TRW	2	HELIOS	SIX-BLADE ASSEMBLY, NONE; FOUR-BLADE ASSEMBLY, LOWER ASSEMBLY SHORTENED
INSULATION	x	x	TRW	2	PIONEERS 10 AND 11	DESIGNED FOR SPECIFIC APPLICATION
HEATERS	x	x	ELECTRO-FILM	2	PROGRAM 169	NONE
ELECTRICAL POWER						
SOLAR ARRAY	x	x	TRW	2	DEFENSE SUPPORT PROGRAM	SIMILAR TO DEFENSE SUPPORT PROGRAM WITH DIFFERENT CONE ANGLE
BATTERY		x	TRW	2	DEFENSE SUPPORT PROGRAM, DSCS II	NEW DESIGN BASED ON DEFENSE SUPPORT PROGRAM AND DSCS II TECHNOLOGY
BATTERY	x		EAGLE PICHER	3	MMC IRAD	Ag-Zn NEW DESIGN; SAME CELLS AS FOR PROBES
POWER CONTROL UNIT	x	x	TRW	2	PIONEERS 10 AND 11	REDESIGN CHARGE/DISCHARGE CONTROLS; MODIFY BUS FILTER/TELEMETRY/COMMAND SLICES
SHUNT RADIATOR	x	x	THERMAL SYSTEM INC.	1	PIONEERS 10 AND 11	NONE
EQUIPMENT CONVERTER	x	x	TRW	2	PIONEERS 10 AND 11	SECONDARY OUTPUT VOLTAGE CIRCUITRY SIMILAR TO PIONEERS 10 AND 11 CTRF
PROPULSION						
PROPELLANT TANKS	x	x	PRESSURE SYSTEMS INC.	1	DSCS-II	DELETE BLADDER; MODIFY MOUNTING ARRANGEMENT
THRUSTERS	x	x	TRW		FLTSATCOM	NONE
PRESSURE TRANSDUCER	x	x	STATHAM INSTRUMENTS	1	PIONEERS 10 AND 11	NONE
TEMPERATURE TRANSDUCER	x	x	ROSEMONT ENGINEERING	1	PIONEERS 10 AND 11	NONE
ROCKET MOTOR (ORBITER ONLY)		x	HERCULES BE-3-A	1	VELA	4.1 PERCENT OFF-LOADING
ATTITUDE CONTROL						
CONTROL ELECTRONICS ASSEMBLY	x	x	TRW	1	PIONEERS 10 AND 11	STELLAR REFERENCE ASSEMBLY LOGIC DELETED; ADD REDUNDANT SUN SENSOR LOGIC; ADD VALVE DRIVER LOGIC
SUN ASPECT SENSOR	x	x	TRW	1	INTELSAT III	MODIFY MASK FIELD OF VIEW FOR IMPROVED ASPECT AND ROLL ACCURACY
CONSCAN PROCESSOR		x	TRW	1	PIONEERS 10 AND 11	IDENTICAL TO PIONEERS 10 AND 11 CONSCAN PROCESSOR EXCEPT CONFIGURE TO ISSUE FIRING PULSES EVERY TWO SPACECRAFT REVOLUTIONS (JUMPER WIRE CHANGE)
STRUCTURE AND MECHANISMS						
STRUCTURE	x	x	TRW	3	NEW	EXISTING SPACECRAFT CONCEPTS
NUTATION DAMPER	x	x	TRW	3	NEW	SIMILAR TO METEOSAT DESIGN
MAGNETOMETER BOOM	x	x	CELESCO	3	NEW	BASED ON VIKING BOOM DESIGN
RELEASE MECHANISMS						
BALL LOCK MECHANISM	x			1	MINUTEMAN PROGRAM	NONE
PIN PULLERS	x		TRW	1	PIONEERS 10 AND 11	NONE

*1 = FLIGHT PROVEN; 2 = DEVELOPED; 3 = NEW

6.3 Mission Reliability

6.3 MISSION RELIABILITY

6.3.1 Probe/Bus Mission Reliability

The success probability for the five system (probe bus, large probe, and three small probes) probe bus mission is dependent on the reliability of each of these systems and the success criteria defined for the large and small probes, respectively. The reliability of the probe bus, the large probe, and each of the small probes is 0.952, 0.945 and 0.964, respectively, for the nominal 110-day mission. The success criteria for the large and small probes have not been determined; therefore, Tables 6-25 and 6-26 have been prepared showing the mission probability for each of the possible success criteria. Figure 6-50 indicates the reliability model used in calculating the mission probabilities for Table 6-25, and Figure 6-51 indicates the reliability model used in calculating the mission probabilities in Table 6-26. These models show that in Table 6-25 the large probe is taken to be mission critical while in Table 6-26 the large probe is taken to be non-mission critical.

The mission time profile for the large probe is 110 days inactive (standby) and 1.1 hour active. However, the piece parts in the isolated, independently powered main bus activation circuit are activated at probe release 25 days before encounter. The impact on reliability of these parts is insignificant since they are backed up by redundant "g" switches.

Table 6-25. Probe Mission Reliability (Large Probe Mission-Critical)

CASE NUMBER	MISSION DEFINITION	RELIABILITY	
		THOR/DELTA	ATLAS/CENTAUR
1	PROBE BUS	0.952	0.952
2	PROBE BUS AND LARGE PROBE	0.920	0.900
3	PROBE BUS, LARGE PROBE, AND AT LEAST ONE SMALL PROBE	0.920	0.900
4	PROBE BUS, LARGE PROBE, AND AT LEAST TWO SMALL PROBES	0.918	0.896
5	PROBE BUS, LARGE PROBE, AND ALL SMALL PROBES	0.842	0.816

MISSION TIMES

BUS: 110 DAYS

LARGE PROBE: 85 DAYS INACTIVE, 25 DAYS COAST, AND 1.1 HOURS ACTIVE.

SMALL PROBES: 89, 93, 97 DAYS INACTIVE; 21, 17, 13 DAYS COAST, RESPECTIVELY, AND 1.1 HOURS INACTIVE.

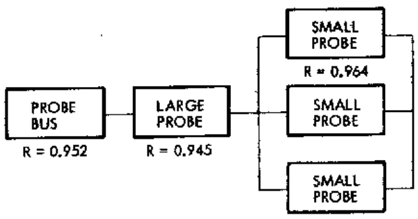
Table 6-26. Probe Mission Reliability (Large Probe Non-Mission-Critical)

CASE NUMBER	MISSION DEFINITION	RELIABILITY	
		THOR/DELTA	ATLAS/CENTAUR
1	PROBE BUS	0.952	0.952
1	PROBE BUS AND AT LEAST ONE SMALL PROBE	0.952	0.952
2	PROBE BUS AND AT LEAST TWO SMALL PROBES	0.950	0.948
3	PROBE BUS AND ALL SMALL PROBES	0.872	0.853

MISSION TIMES

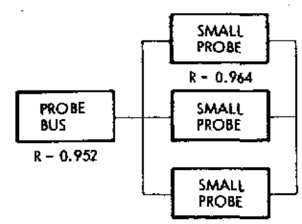
BUS: 85 TO 110 DAYS

SMALL PROBES: 89, 93, 97 DAYS INACTIVE; 21, 17, 13 DAYS COAST, RESPECTIVELY, AND 1.1 HOURS INACTIVE.



SMALL PROBE RELIABILITY CONFIGURATION IS DEPENDENT ON SUCCESS CRITERIA - FROM ONE OF THREE REQUIRED TO ALL THREE REQUIRED (SEE TABLE 6-25)

Figure 6-50. Atlas/Centaur Probe Mission Reliability Diagram (Probe Bus and Large Probe Mission Critical)



SMALL PROBE RELIABILITY CONFIGURATION IS DEPENDENT ON SUCCESS CRITERIA - FROM ONE OF THREE REQUIRED TO ALL THREE REQUIRED (SEE TABLE 6-26)

Figure 6-51. Atlas/Centaur Probe Mission Reliability Diagram (Probe Bus Only Mission Critical)

The small probe mission profile for Tables 6-25 and 6-26 is 110 days inactive (standby) and 1.1 hour active. The three small probes each have the same type timing circuit as the large probe with backup "g" switches and are released 21, 17, and 13 days prior to entry, respectively. Again the timing circuits have insignificant impact on reliability due to redundancy.

Tradeoff studies and launch vehicle impact are discussed in the probe and probe bus sections that follow.

6.3.2 Probe Reliability ALL PROBE CONFIGURATIONS

The reliability study tasks involved: identification of critical items/ single failure points; reliability versus failure resource resolution studies (weight and volume); and active participation during the system configuration trades via reliability assessments, as noted in Figures 6-52 and 6-53. Failure mode and effects analysis efforts are noted in Appendix 6I. These efforts were instrumental in guiding the probe design to produce the highest reliability at minimum weight and volume resource costs. The achieved resource cost effective basis is indicated by the fact that further failure resource resolution expenditures are large, likelihoods of failures are small, and reliability gains are insignificant.

The reliability of the electrical power subsystem received major attention because of its critical nature and the weak link status of the battery. Analysis was made of the Pioneer 10 and 11 type active battery cell bypass circuitry, passive battery cell bypass diodes, 13-cell battery with external electronic discharge regulation, and battery cell out (short) capability. The leakage current of the Pioneer 10 and 11 type active battery cell bypass circuitry demands power throughout the mission (costing significant weight) and was therefore deleted from further consideration. The

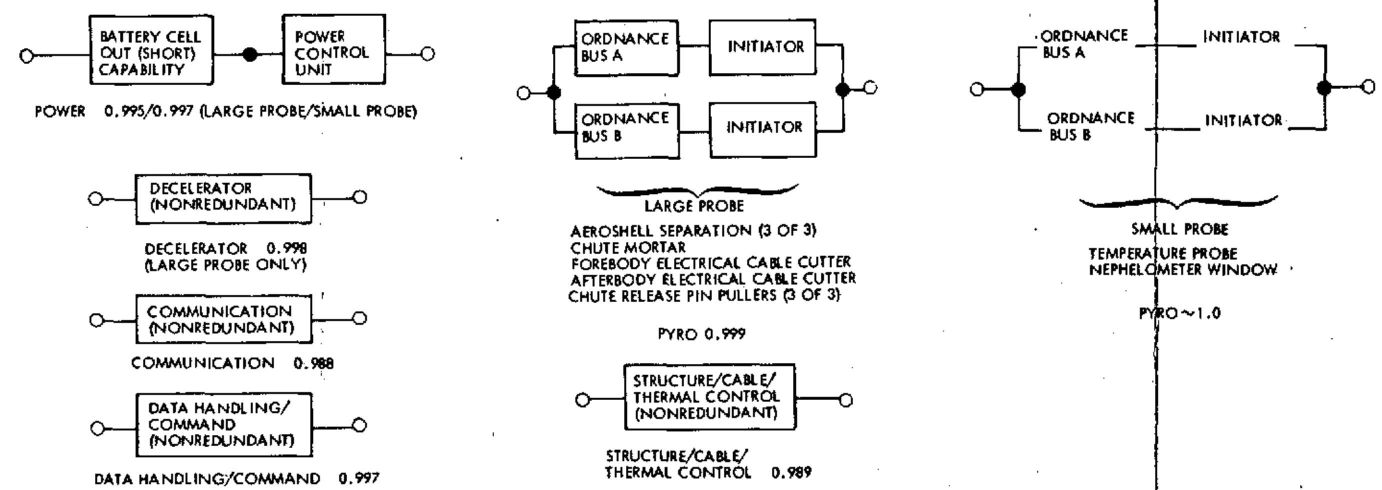
SINGLE THREAD RELIABILITY

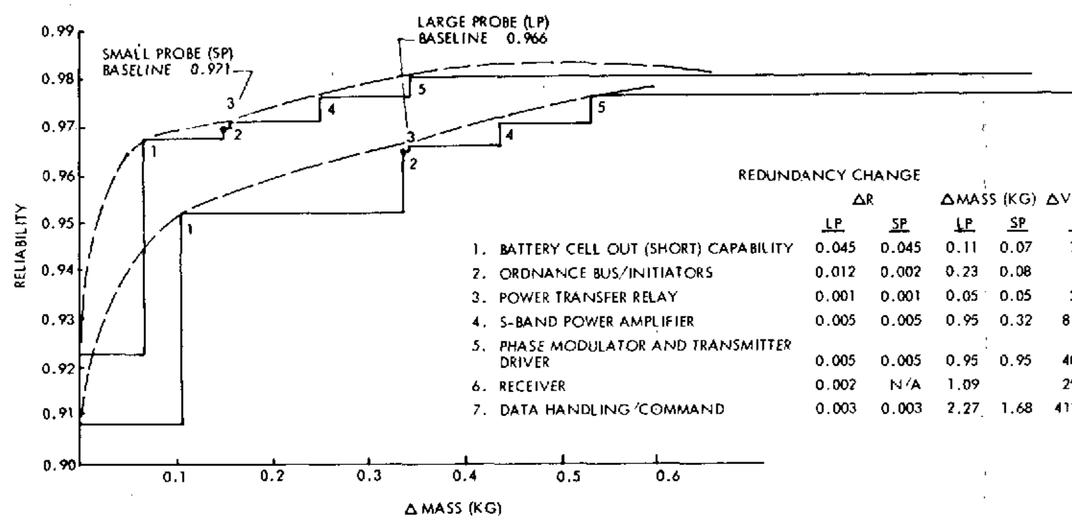
LARGE PROBE	SMALL PROBE
0.908	0.923

BASELINE RELIABILITY

SUBSYSTEM	LARGE PROBE	SMALL PROBE
POWER	0.995	0.997
PYRO	0.999	~1.0
DECELERATOR	0.998	N/A
COMMUNICATION	0.988	0.988
DATA HANDLING/ COMMAND	0.997	0.997
STRUCTURE/CABLE/ THERMAL CONTROL	0.989	0.989
	0.966	0.971

(ESTIMATES, RELIABILITY VERSUS WEIGHT, BLOCK DIAGRAMS, AND SINGLE FAILURE POINTS)





REDUNDANCY CHANGE

	ΔR LP	ΔR SP	ΔMASS (KG) LP	ΔMASS (KG) SP	ΔVOLUME (CM^3) LP	ΔVOLUME (CM^3) SP
1. BATTERY CELL OUT (SHORT) CAPABILITY	0.045	0.045	0.11	0.07	73.74	40.97
2. ORDNANCE BUS/INITIATORS	0.012	0.002	0.23	0.08	1.97	0.66
3. POWER TRANSFER RELAY	0.001	0.001	0.05	0.05	20.48	20.48
4. S-BAND POWER AMPLIFIER	0.005	0.005	0.95	0.32	819.36	131.10
5. PHASE MODULATOR AND TRANSMITTER DRIVER	0.005	0.005	0.95	0.95	409.68	409.68
6. RECEIVER	0.002	N/A	1.09		294.97	
7. DATA HANDLING/COMMAND	0.003	0.003	2.27	1.68	4129.55	1982.84

SINGLE FAILURES

CATASTROPHIC SINGLE FAILURES (TOTAL LOSS OF MISSION)	CRITICAL SINGLE FAILURES (65-90% LOSS OF MISSION)
POWER/ORDNANCE BATTERY (CELL OPEN) INTERFACE CABLE CUTTER (TRW SIDE OF INTERFACE) COMMUNICATION PHASE MODULATOR AND TRANSMITTER DRIVER S-BAND POWER AMPLIFIER DIPLEXER (a) ANTENNA DATA HANDLING/COMMAND TIMING GENERATOR FORMAT GENERATOR DATA STORAGE UNIT DC/DC CONVERTER (a) LARGE PROBE ONLY (EQUIPMENT NOT APPLICABLE TO SMALL PROBE)	POWER/ORDNANCE BATTERY HEATER POWER CONTROL UNIT SAFE/ARM RELAY (b) DECELERATOR CHUTE MORTAR (a) AEROSHELL SEPARATION NUT SEPARATOR DEVICES (3) (a) CHUTE RELEASE PIN PULLERS (3) (a) AEROSHELL FOREBODY ELECTRICAL CABLE CUTTER (a) BASE COVER ELECTRICAL CABLE CUTTER (a) COMMUNICATION RECEIVER (a) DATA HANDLING/COMMAND MULTIPLEXER SIGNAL CONDITIONING A/D CONVERTER BIPHASE MODULATOR AND CONVOLUTIONAL ENCODER (b) LARGE PROBE ONLY
REDUNDANCY POWER/ORDNANCE BATTERY CELL OUT (SHORT) CAPABILITY POWER CONTROL UNIT POWER TRANSFER RELAYS PYRO TRIGGERING CIRCUITS PYRO BUSES PYRO INITIATORS DATA HANDLING/COMMAND TIMING GENERATOR G SWITCHES	

Figure 6-52. Thor/Delta Reliability Studies

SINGLE THREAD RELIABILITY

LARGE PROBE	SMALL PROBE
0.888	0.915

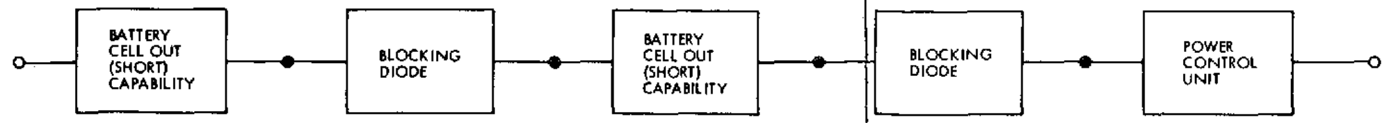
BASELINE RELIABILITY

SUBSYSTEMS	LARGE PROBE	SMALL PROBE
POWER	0.983	0.990
PYRO	0.999	~1.0
DECELERATOR	0.996	N/A
COMMUNICATION	0.983	0.991
DATA TRANSMISSION UNIT	0.994	0.994
STRUCTURE/CABLE/ THERMAL CONTROL	0.989	0.989
	0.945	0.964

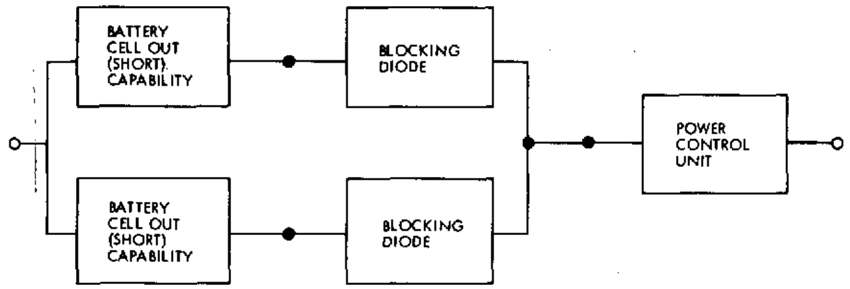
(ESTIMATES, RELIABILITY VERSUS WEIGHT, BLOCK DIAGRAMS, AND SINGLE FAILURE POINTS)

◇ A/C III

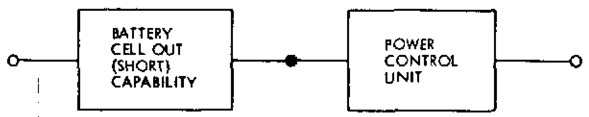
SUBSYSTEM RELIABILITY BLOCK DIAGRAMS



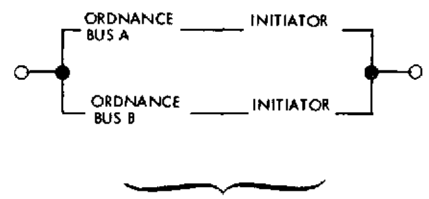
LARGE PROBE POWER (TOTAL SUCCESS) 0.983



LARGE PROBE POWER (50% MISSION) 0.986



SMALL PROBE POWER 0.990

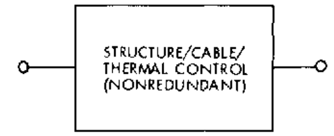


SMALL PROBE
TEMPERATURE PROBE
NEPHELOMETER WINDOW

PYRO ~ 1.0

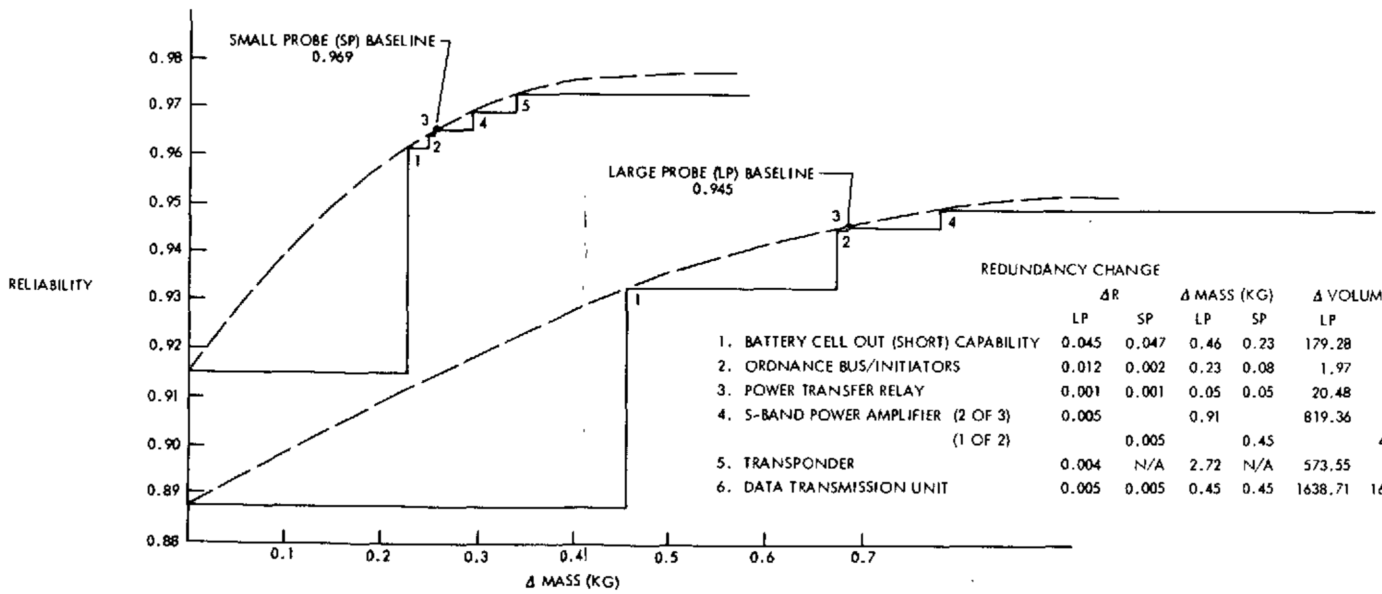
COMMUNICATION 0.983/0.991
(LARGE PROBE/SMALL PROBE)

DATA TRANSMISSION UNIT 0.994



STRUCTURE/CABLE/THERMAL CONTROL 0.989

DECELERATOR 0.996



REDUNDANCY CHANGE

	ΔR LP	ΔR SP	Δ MASS (KG) LP	Δ MASS (KG) SP	Δ VOLUME (CM^3) LP	Δ VOLUME (CM^3) SP
1. BATTERY CELL OUT (SHORT) CAPABILITY	0.045	0.047	0.46	0.23	179.28	89.64
2. ORDNANCE BUS/INITIATORS	0.012	0.002	0.23	0.08	1.97	0.66
3. POWER TRANSFER RELAY	0.001	0.001	0.05	0.05	20.48	20.48
4. S-BAND POWER AMPLIFIER (2 OF 3)	0.005		0.91		819.36	
(1 OF 2)		0.005		0.45		409.68
5. TRANSPONDER	0.004	N/A	2.72	N/A	573.55	N/A
6. DATA TRANSMISSION UNIT	0.005	0.005	0.45	0.45	1638.71	1638.71

SINGLE FAILURES

CATASTROPHIC SINGLE FAILURES
(TOTAL LOSS OF MISSION)

POWER/ORDNANCE
- BATTERY (CELL OPEN) (a)
- INTERFACE CABLE CUTTER (TRW SIDE OF INTERFACE)

COMMUNICATION
- TRANSPONDER MODULATOR DRIVER
- S-BAND POWER AMPLIFIER (TWO LARGE PROBE, ONE SMALL PROBE)
- HYBRID COUPLERS (b)
- DIPLEXER
- ANTENNA
- PHASE MODULATOR AND TRANSMITTER DRIVER (a)

DATA TRANSMISSION UNIT

(a) SMALL PROBE ONLY
(b) LARGE PROBE ONLY (EQUIPMENT NOT APPLICABLE TO SMALL PROBE)
(c) LARGE PROBE ONLY

CRITICAL SINGLE FAILURES
(65-90% LOSS OF MISSION)

POWER/ORDNANCE
- BATTERY HEATER
- POWER CONTROL UNIT SAFE/ARM RELAY (c)
- PILOT CHUTE MORTAR (b)
- AEROSHELL SEPARATION
  - NUT SEPARATOR DEVICES (3) (b)
- AFTERBODY/MAIN CHUTE SEPARATION
  - NUT SEPARATOR DEVICES (3)

COMMUNICATION
- TRANSPONDER RECEIVER (b)

DECELERATOR
- MAIN CHUTE (b)

REDUNDANCY

POWER/ORDNANCE
- BATTERY CELL OUT (SHORT) CAPABILITY
- POWER CONTROL UNIT
  - POWER TRANSFER RELAYS
  - PYRO TRIGGERING CIRCUITS
- PYRO BUSES
- PYRO INITIATORS

DATA HANDLING/COMMAND
- TIMING GENERATOR G SWITCHES

Figure 6-53. Atlas/Centaur Reliability Studies

passive battery cell bypass diodes represented an expenditure of 0.13 kg and 98.32 cm^3 per battery for an insignificant reliability increase and were deleted from further consideration. The 13-cell battery with external electronic discharge regulation represented an expenditure of 7.62 kg for the large probe and 3.23 kg for the small probes with lesser reliability than the baselined battery configuration. The baselined probe configurations incorporate batteries of 20 cells sized with margin such that external electronic discharge regulation is unnecessary and any cell short failure does not compromise mission success.

The mission success considerations centered on the stated scientific objectives of the probe mission. The NASA Ames Research Center "Requirements for Pioneer Venus Mission Systems Design Study" and "Statement of Work" defined the scientific objectives of the multiple-probe mission as investigation of the dense regions of the Venus atmosphere to determine the:

1) Nature and composition of the clouds (100 km to surface);
2) Composition and structure of the atmosphere to the surface ($>$ 100 km to the surface);
3) General circulation pattern of the lower atmosphere;
4) Characteristics of the planetary environment interaction with the interplanetary medium and by cursory investigation, the composition and structure of the ionosphere ($>$ 100 km $<$ 500 km).

These mission objectives were considered in the light of the Version III (December 1973) probe nominal payloads. These considerations led to the instrument relationships and contributions to mission objectives per Table 6-27.

These considerations, relationships, and contributions indicate Mission Objective Success satisfactions due to obtainments of the various autonomous instruments/probes. These contribution percentages were based on value estimates to the mission objectives and the projected degree of contribution to the mission objective should a given instrument on a given probe be obtained. The major contribution of a figure of this type lies in the indication of potentials for partial mission objectives successes (with/without given objectives 1, 2, 3, 4--with/without given probes--small probe 1, small probe 2, small probe 3, large probe). Whereas, a standard figure would portray only total success (all objectives and all probes). These con-

Table 6-27. Version III Nominal Payload Instruments Related to Objectives

ALL VERSION III SCIENCE PAYLOAD

INSTRUMENT	OBJECTIVE 1 RELATION	OBJECTIVE 1 CONTRIBUTION	OBJECTIVE 2 RELATION	OBJECTIVE 2 CONTRIBUTION	OBJECTIVE 3 RELATION	OBJECTIVE 3 CONTRIBUTION	OBJECTIVE 4 RELATION	OBJECTIVE 4 CONTRIBUTION
LARGE PROBE								
PRESSURE, TEMPERATURE	D	24%	D	24%	D	24%	I	
ACCELEROMETERS	I		D	15%	I		D	15%
MASS SPECTROMETER	D	10%	D	15%	I		I	
CLOUD PARTICLE ANALYZER	D	5%	N		I		N	
SOLAR FLUX RADIOMETER	D	5%	I		I		N	
PLANETARY FLUX DETECTOR	D	5%	I		I		N	
AUREOLE/EXTINCTION DETECTOR	D	5%	I		D	5%	N	
TRANSPONDER	I		I		D	15%	I	
NEPHELOMETER	D	10%	N		I		N	
HYGROMETER	D	3%	D	2%	I		N	
SHOCK LAYER RADIOMETER	I		D	5%	N		D	5%
SMALL PROBES (3)								
PRESSURE, TEMPERATURE	D	8%	D	8%	D	8%	I	
NEPHELOMETER	D	3%	N		I		N	
ACCELEROMETER	I		D	5%	I		D	5%
MAGNETOMETER	N		N		N		D	#1 50% #2 10% #3 5%
STABLE OSCILLATOR	N		I		D	#1 15% #2 10% #3 7%	N	
TOTAL		100%		100%		100%		100%

D: DIRECTLY RELATED
I: INDIRECTLY RELATED
N: LITTLE OR NO RELATION

T/D III
31 W
T/D III

siderations also involved the probe's Venus encounter mission phases/events and the state of probe science, systems, and timer/programmer as tabulated:

VENUS ENCOUNTER MISSION PHASES/EVENTS

1 BUS/LARGE PROBE AND SMALL PROBE SEPARATIONS
SCIENCE - OFF TIMER/PROGRAMMER - ON
SYSTEMS - OFF

2 PRE-PLANET ATMOSPHERIC ENTRY
CHECKOUT AND PREPARATION
SCIENCE - ON ACQUIRE DATA
SYSTEMS - ON

3 ENTRY
ACQUIRE DATA TRANSMIT DATA
STORE DATA

4 HIGH ALTITUDE DECELERATION
STORE ACCELEROMETER DATA
IN BLACKOUT MEMORY

5 UNCOVER INSTRUMENTS/EXPERIMENTS

6 HIGH ALTITUDE DESCENT
ACQUIRE DATA TRANSMIT DATA
STORE DATA

7 VENUS DESCENT
ACQUIRE DATA TRANSMIT DATA
STORE DATA

8 LOW ALTITUDE DESCENT
ACQUIRE DATA TRANSMIT DATA
STORE DATA

VENUS ENCOUNTER — 1 — 2 — 3 — 4 — 5 — 6 — 7 — 8 — VENUS SURFACE IMPACT

T/D III
31 W T/D III

The consideration of these mission objectives disclosed their multifaceted nature due to the interest in atmospheric structure, composition and distribution. These interests dictate wide latitude and longitude displacement of probe targeting with several different sets investigated as noted in paragraph 4.2.2. Thus, in consideration of the mission objectives and the probe targeting displacements, the probes should be considered both complementary and supplementary.

The conclusion to be drawn from these efforts is that the individual instrument large probe/small probe associations with relationships to the mission objectives means that each of the mission objectives can be individually satisfied to some degree despite loss of any instrument(s) and/or probe(s).

These same efforts were undertaken for the Version IV (April 1974) probe nominal payloads and are evidenced in Table 6-28 (similar to Table 6-27). To supplement these tables, Figure 6-54 portrays the mission objective data contribution of the large and small probes.

Table 6-28. Version IV Nominal Payload Instruments Related to Objectives

ALL VERSION IV SCIENCE PAYLOAD

INSTRUMENT	OBJECTIVE							
	RELATION/CONTRIBUTION		RELATION/CONTRIBUTION		RELATION/CONTRIBUTION		RELATION/CONTRIBUTION	
LARGE PROBE	1		2		3		4	
PRESSURE, TEMPERATURE	D	24%	D	24%	D	24%	I	
ACCELEROMETERS	I		D	15%	I		D	55%
NEUTRAL MASS SPECTROMETER	D	10%	D	15%	I		I	
CLOUD PARTICLE SIZE ANALYZER	D	3%	N		I		N	
SOLAR RADIOMETER	D	5%	I		I		N	
IR FLUX RADIOMETER	D	4%	I		I		N	
HYGROMETER	D	3%	D	2%	I		N	
WIND ALTITUDE RADAR	I		I		D	15%	N	
GAS CHROMATOGRAPH	D	3%	D	5%	I		I	
SMALL PROBES								
PRESSURE, TEMPERATURE	D	8%	D	8%	D	8%	I	
NEPHELOMETER	D	4%	N		I		N	
ACCELEROMETER	I		D	5%	I		D	15%
STABLE OSCILLATOR	N		1		D	#1 15% #2 12% #3 10%	N	
IR FLUX DETECTOR	D	4%	I		I		N	
TOTAL		100%		100%		100%		100%

D: DIRECTLY RELATED
I: INDIRECTLY RELATED
N: LITTLE OR NO RELATION

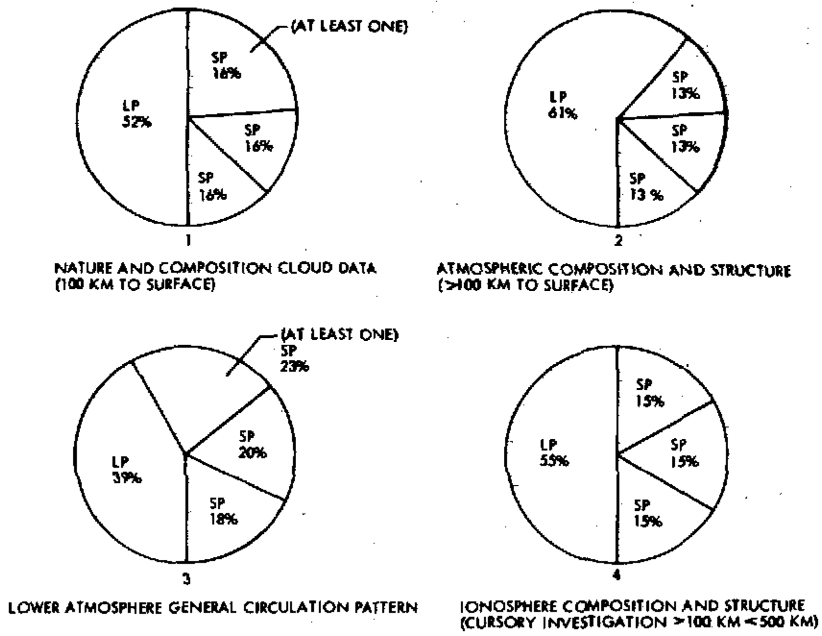


Figure 6-54. Mission Objective Data Contribution Large Probe/Small Probe

6.3.2.1 Probe Reliability, Thor/Delta T/D III T/D III

The estimated reliability of the baselined Thor/Delta large and small probe is 0.966 and 0.971, respectively. Details for these estimates are given in Figure 6-52. The baselined configurations represent weight/volume resource expenditures of 0.39 kg/96.19 cm^3 for the large probe and 0.20 kg/62.11 cm^3 for the small probe to incorporate redundancy. This resource expenditure was used to provide battery cell out (short) capability; redundant power transfer relays, pyrotechnic initiators, and g switches; and bus/probe power isolation diodes. Additional expenditure of failure resolving weight/volume resources could only be justified by significant contribution to mission success. The baselined probe configurations are such that the remaining failures are very unlikely and thus no significant contribution to mission success remains.

Power

Power Control Unit (Safe and Arm Relay)	This unit is fairly costly in expenditure of resolution resources as three relays would be required to assure failure free arm and safe functions (delta 0.10 kg/40.96 cm^3). With small probability of failure (5/10 000), such expenditure is unjustified.

Data Handling and Command	This unit is highly costly in expenditure of resolution resources. However, redundancy would take care of three catastrophic single failure functions and must be considered. Active data handling command redundancy is precluded because of multiple voting fault isolation and switching, additional power, and additional thermal requirements penalties. Thus, only a standby data handling command with selection opted at preseparation checkout and no redundancy after separation would be feasible. The resource penalties are excessive to resolve a 3/1000 chance of failure.
Communication	Each single failure point unit, except the antenna, could be eliminated by standby block redundancy. However, resource penalties are large and RF power loss/switching considerations would have to be resolved. Expenditure of resources to correct these units with small likelihood of failure is unjustified.
Decelerator	The decelerator subsystem and each of its elements comprise single failure points. However, a single chute system is the simplest, lightest, and most cost-effective configuration that can be adapted and possesses sufficient reliability to justify same. Whereas, completely redundant and multiple parachute systems involve entanglement problem considerations.

A/C III A/C III

6.3.2.2 Probe Reliability, Atlas/Centaur

A/C IV A/C IV

From a reliability standpoint, the Thor/Delta and Atlas/Centaur probes were basically alike. The primary difference involved the electrical power batteries and communication power amplifiers. In the interest of commonality, the Atlas/Centaur large probe used two each of the small probe batteries and power amplifiers, both being necessary for total mission success. These batteries were configured of 20 cells sized with margin such that (1) external discharge regulation is unnecessary, (2) any cell short failure did not compromise mission success, and (3) any cell open failure in the large probe was not catastrophic with another battery remaining. In addition, the large probe batteries could be considered redundant for 50 percent mission success.

The estimated reliability of the baselined Atlas/Centaur large probe is 0.945/0.948 (total/50% mission). The estimated reliability of the baselined Atlas/Centaur small probe is 0.964. Details for these estimates are given in Figure 6-53. The baselined configurations represent weight/volume resource expenditures of 0.84 kg/201.73 cm^3 for the large probe and 0.46 kg/110.78 cm^3 for the small probe to incorporate redundancy. This resource expenditure was used to provide battery cell out (short) capability, redundant power transfer relays, pyrotechnic initiators, and g switches, and bus/probe power isolation diodes.

The Thor/Delta position regarding additional expenditure of volume/weight resources is discussed in paragraph 6.3.2.1. The same rationale, except for the decelerator subsystem, is applicable for Atlas/Centaur. The decelerator subsystem on the Atlas/Centaur large probe is a two-chute system (pilot and main) due to the required main chute size and each of its elements comprise single failure points. However, a single pilot chute and single main chute is the simplest, lightest, and most cost effective configuration that can be adapted for a two-chute system and possesses sufficient reliability to justify same. Whereas, completely redundant and multiple pilot and main chutes would involve entanglement problem considerations.

6.3.3 Probe Bus Reliability ALL PROBE CONFIGURATIONS

6.3.3.1 Tradeoff Studies T/D III

For the Thor/Delta launch configuration, a weight/reliability tradeoff study was performed to optimize the probe bus within launch vehicle weight constraints. The study used a programming technique that considers all possible combinations of weight and reliability. The criteria for the optimum design for the probe bus was minimizing single-failure points and judicially adding redundancy to maximize reliability improvement per unit of additional weight. Figure 6-55 plots reliability versus weight for optimum redundancy configurations, with optimum configurations defined as those providing the greatest increase in reliability for the additional weight of redundant units. The redundancy configuration for each labeled point on the curve is shown in Table 6-29.

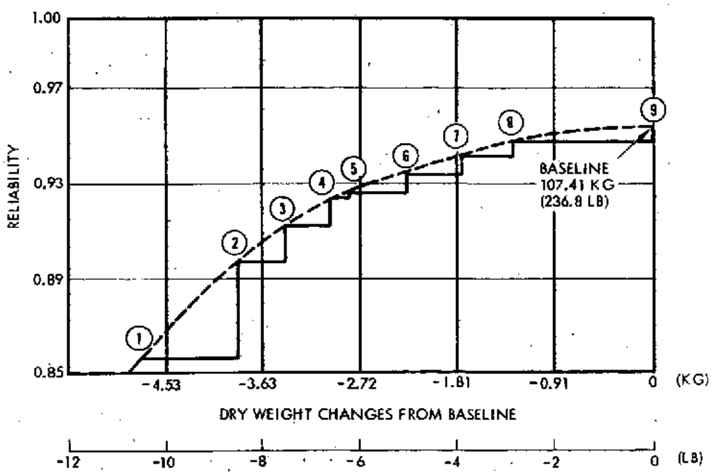


Figure 6-55. Reliability/Weight Tradeoff Curve for Thor/Delta Probe Bus

Table 6-29. Redundancy Configurations Based on Optimization Curve

OPTIMIZATION CURVE REFERENCE	INCORPORATED REDUNDANCY							
	TRANSMITTER	RECEIVER	S-BAND AMPLIFIER	DSL/SPC	SUN SENSOR	DTU	DDU	CDU
1	← NONREDUNDANT →							
2						X		
3				X		X		
4				X		X	X	
5				X	X	X	X	
6	X			X	X	X	X	
7	X	X		X	X	X	X	
8	X	X	X	X	X	X	X	
9	X	X	X	X	X	X	X	X

6.3.3.2 Thor/Delta Configuration

The reliability block diagram for the probe bus Thor/Delta configuration is shown in Figure 6-56. The key reliability features are listed in Figure 6-57.

The communication subsystem configuration is redundant except for those items which cannot be made redundant, i.e., hybrids, transfer switches, etc. The receivers are in active redundancy and each has a dedicated converter. The transmitters are standby redundant with each transmitter having a dedicated converter. The transmitter power amplifiers are in standby redundancy. Although there are several antennas on the spacecraft, they are not considered redundant since each is used for specific purposes during the mission.

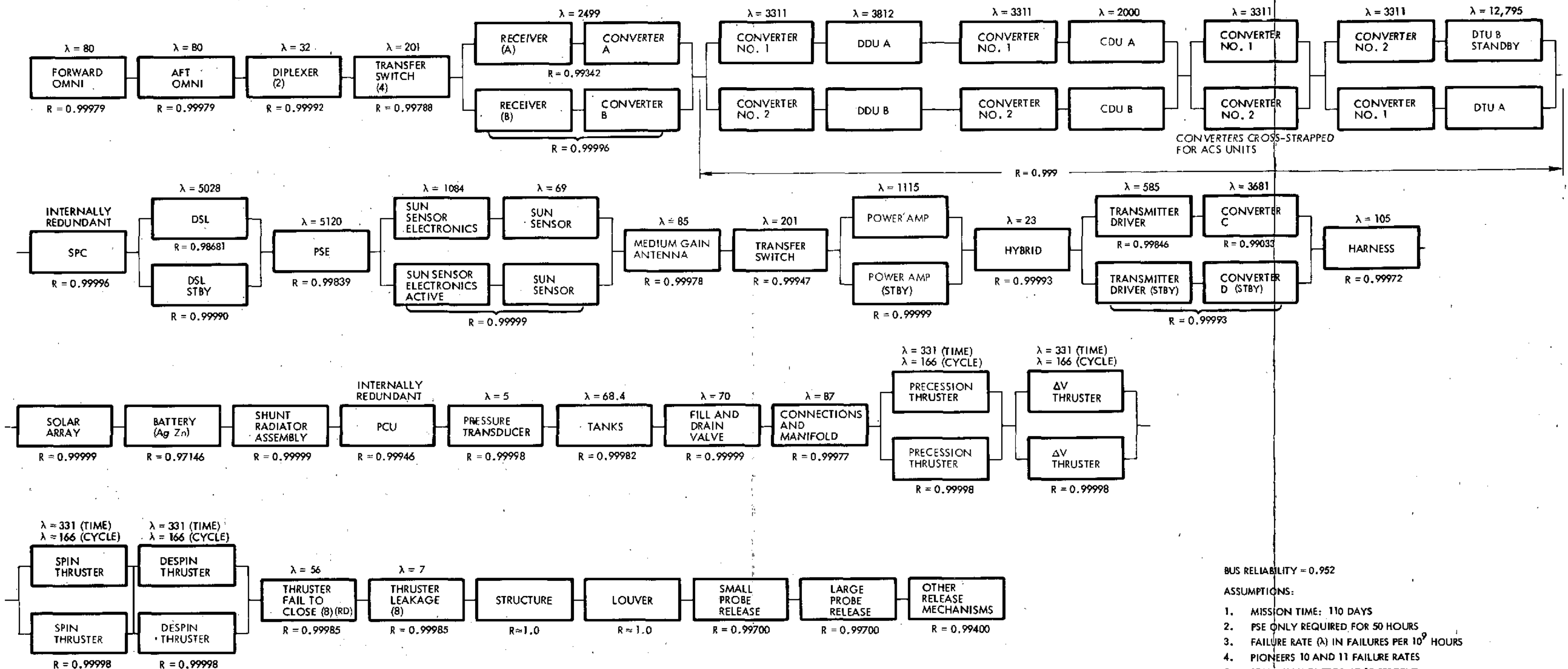
The command and data handling subsystem is primarily based on the Pioneers 10 and 11 design. The digital decoder units (DDU) and command distribution unit (CDU) employ active redundancy; the digital telemetry unit (DTU) incorporates standby redundancy. Each unit receives power from one of the two central converters with no redundant units receiving power from the same converter.

The attitude control subsystem is completely redundant except for the program storage and execution subassembly (PSE) which is required for only 50 hours during the mission. The PSE functions, if necessary, can be performed by ground command. The sensor and power control subassembly (SPC) is internally redundant and receives cross-strapped power from the redundant central converters with isolation provided within the SPC. The duration and steering logic subassemblies (DSL) are in standby redundancy and the sun sensor electronics and sensors are internally redundant. All units receive power from the cross-strapped redundant central converters through the SPC.

The electrical power subsystem is designed for maximum reliability within the weight constraints and requirements of the system. The solar array has a 0.99999 probability of supplying 96.6 percent of the available power which is 1.6 percent greater than the requirement for 95 percent of the available power. The 13-cell silver-zinc battery retains commonality in design between probe bus and the small probe battery design. The battery is required for approximately one hour near earth and for the rest of the mission it will be open-circuited and commanded on-line only during pulse load requirements. The shunt radiator has built-in redundancy and the power control unit (PCU) is internally redundant.

The propulsion subsystem has redundant thrusters and valve drivers for precession, ΔV, spin, and despin. A failure of the tank pressure transducer is not mission critical as long as it does not fail in the leakage mode, since propellant usage can be analytically determined. The propulsion tanks and lines are single-point failures.

The structural/thermal subsystem can tolerate one failed louver without causing thermal imbalance. The deployment mechanisms for the large probe are ball lock devices which have demonstrated high reliability on the Minuteman program. However, each of the ball locks will be activated by redundant squibs, deviating from the gas generator approach



BUS RELIABILITY = 0.952

ASSUMPTIONS:

1. MISSION TIME: 110 DAYS
2. PSE ONLY REQUIRED FOR 50 HOURS
3. FAILURE RATE (λ) IN FAILURES PER 10^9 HOURS
4. PIONEERS 10 AND 11 FAILURE RATES
5. STANDBY K FACTOR AT 10 PERCENT

Figure 6-56. Thor/Delta Probe Bus Reliability Block Diagram

T/D III

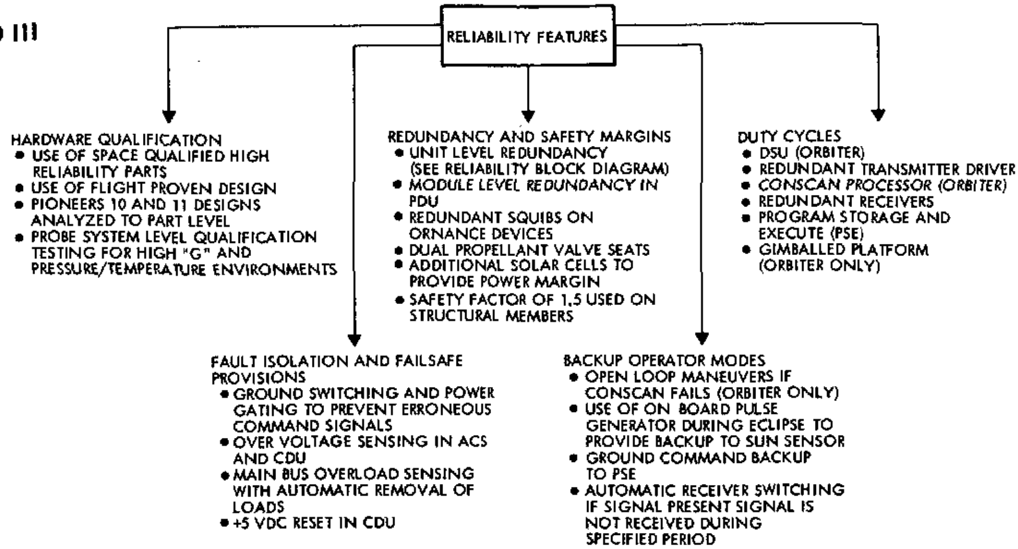


Figure 6-57. Key Reliability Features

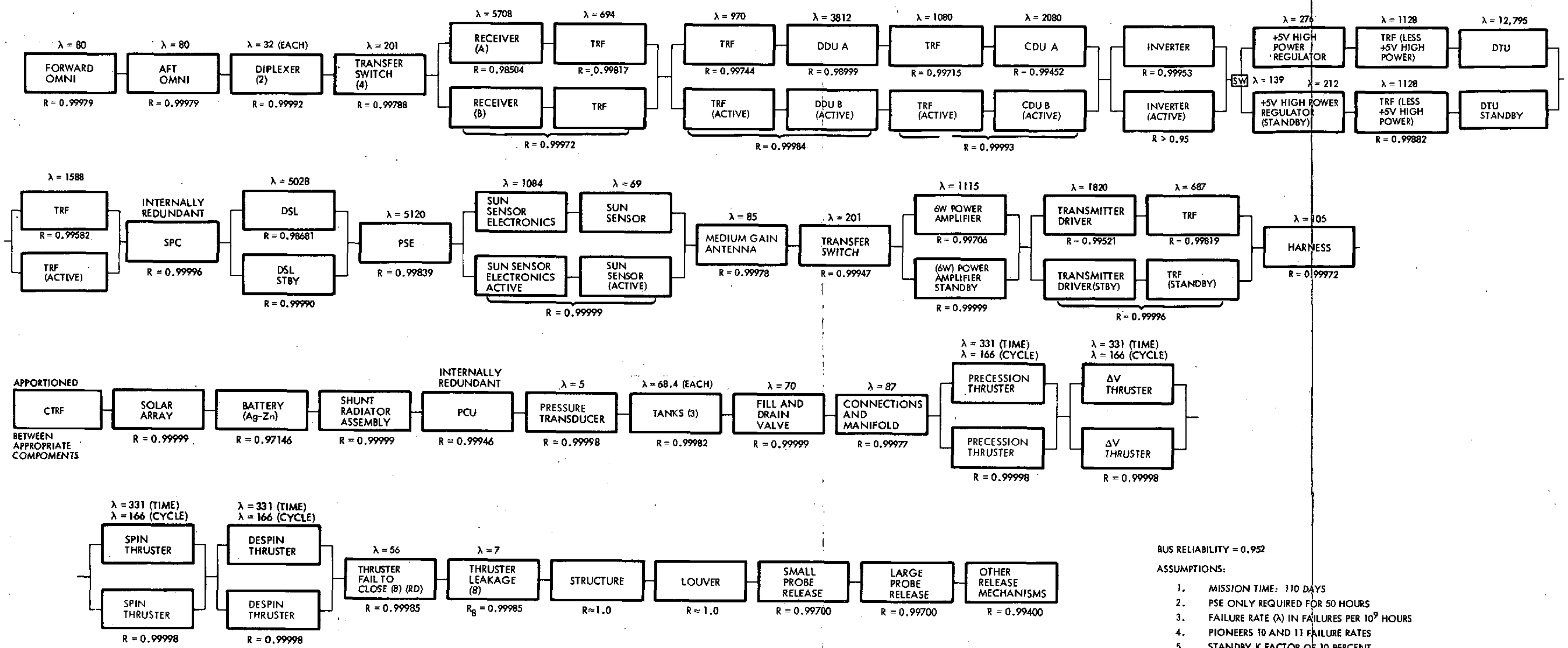
used on Minuteman. This is not believed to adversely affect the reliability. Redundant firing circuits, identical to those on Pioneers 10 and 11, will be used to activate the squibs. The umbilical from the probe bus to the large probe will be severed by a cable cutter having redundant squibs activated by redundant firing commands. The small probes will be released by two pin pullers sequentially fired to relax the strain energy prior to deployment. Each pin puller has redundant squibs activated by redundant firing commands. The small probe umbilicals will be severed like the large probe umbilical.

The system level failure modes and effect analyses (FMEA) is presented in Appendix 6.3I and identifies each of the system level single-point failures.

6.3.3.3 Atlas/Centaur Configuration A/C III

The reliability block diagrams and model for the preferred Atlas/Centaur configuration is presented in Figure 6-58. The reliability criteria used to optimize the design was the elimination of single-point failures. No reliability versus weight optimization analysis was performed since the Atlas/Centaur configuration is not weight constrained.

From a reliability standpoint the only changes between the Thor/Delta and the Atlas/Centaur configurations occur in the electrical power and communication subsystems. In the electrical power subsystem, the



BUS RELIABILITY = 0.952

ASSUMPTIONS:

1. MISSION TIME: 110 DAYS
2. PSE ONLY REQUIRED FOR 50 HOURS
3. FAILURE RATE (λ) IN FAILURES PER 10^9 HOURS
4. PIONEERS 10 AND 11 FAILURE RATES
5. STANDBY K FACTOR OF 10 PERCENT

Figure 6-58. Atlas/Centaur Probe Bus Reliability Block Diagram

FOLDOUT FRAME 1

FOLDOUT FRAME 2

A/C III DC-to-DC converters are replaced by an inverter and central-transformer-rectifier-filter (CTRF) as utilized on Pioneers 10 and 11. Redundant independently isolated secondary power will be provided to each of the redundant units except for the attitude control subsystem, which receives redundant cross-strapped secondary power. The change in the communication subsystem is that the receivers and transmitter drivers with dedicated converters used on the Thor/Delta configuration will be replaced by Pioneers 10 and 11 units. These receivers and transmitters require secondary power which is provided by additional CTRF outputs. However, even with these changes the probe bus mission reliability for the Atlas/Centaur and Thor/Delta configurations is approximately the same.

The FMEA is presented in Appendix 6.3I and identifies each of the system level single-failure points.

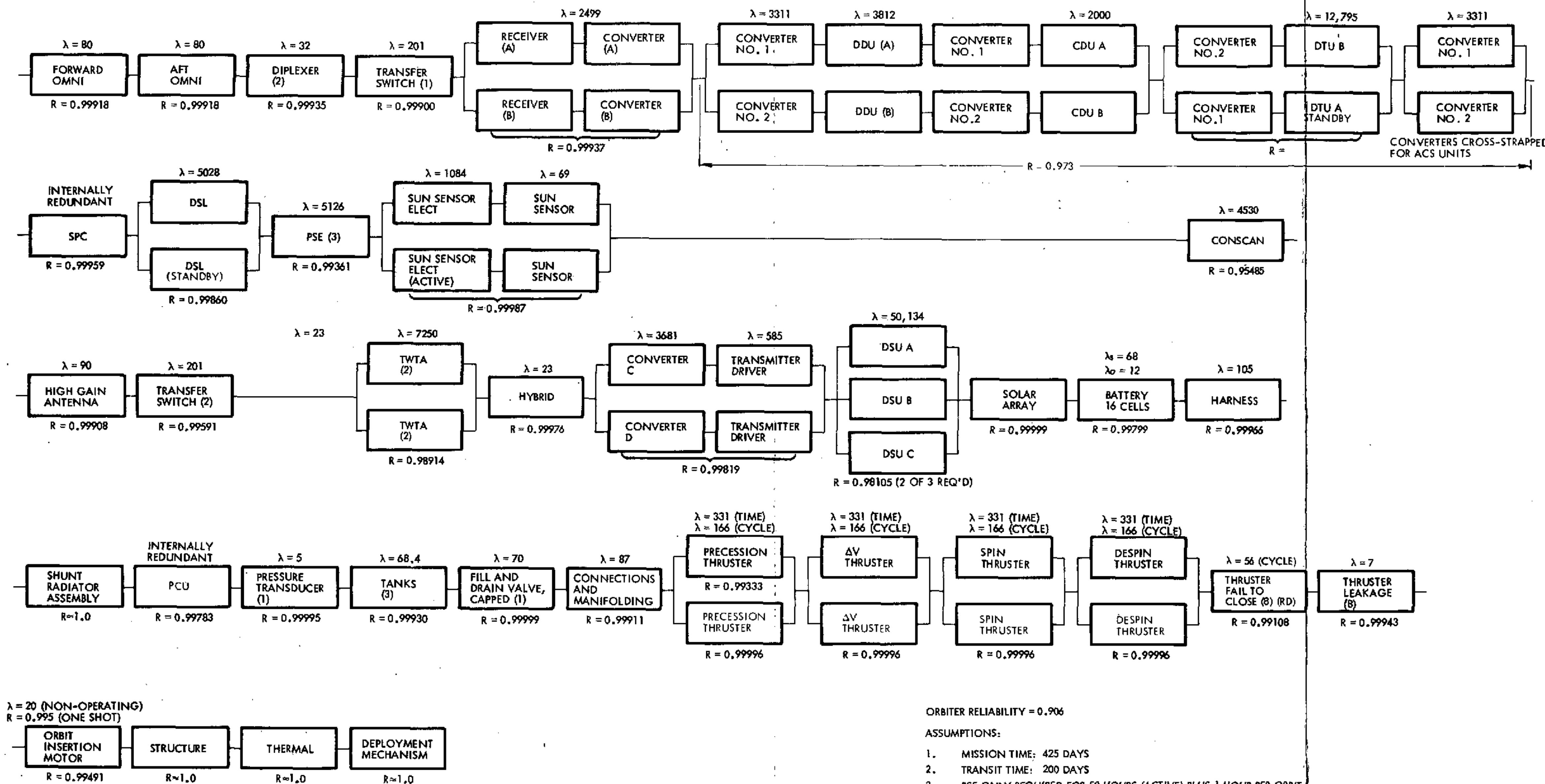
6.3.4 Orbiter Reliability ALL ORBITER CONFIGURATIONS

The orbiter mission is for 425 days and mission reliabilities are 0.906 and 0.925 for the Thor/Delta and Atlas/Centaur launch vehicle configurations, respectively. The transit time for each case was assumed to be 200 days with the remaining 225 days in orbit. Each orbiter design is derived from the related probe bus design with minor configuration changes in the electrical power, data handling, and communication subsystems to accommodate unique mission requirements. The basic weight versus reliability optimization performed for the Thor/Delta configuration is valid for the orbiter to determine redundancy levels since the same basic equipment is utilized.

6.3.4.1 Thor/Delta Configuration T/D III

The reliability block diagram for the Thor/Delta configuration is shown in Figure 6-59. The Thor/Delta orbiter reliability model includes the following changes to the probe bus design that impact reliability.

- The electrical power subsystem has a different solar array design which has a 0.99999 probability of delivering 97 percent of the available array power (95 percent of available power is required). A nickel-cadmium battery replaces the silver-zinc type. During the first part of the Venus orbit mission the battery experiences a 30-percent depth of discharge. During this period there is very low probability of losing the battery due to a cell short. After 165 days of the Venus orbit mission, and for a period of 25 days, a very high depth of discharge occurs (80%)



ORBITER RELIABILITY = 0.906

ASSUMPTIONS:

1. MISSION TIME: 425 DAYS
2. TRANSIT TIME: 200 DAYS
3. PSE ONLY REQUIRED FOR 50 HOURS (ACTIVE) PLUS 1 HOUR PER ORBIT
4. THE CONSCAN IS CONSIDERED MISSION NONCRITICAL
5. FAILURE RATE (λ) IN FAILURES PER 10^9 HOURS
6. PIONEER 10 AND 11 FAILURE RATES
7. STANDBY K FACTOR OF 10 PERCENT

Figure 6-59. Thor/Delta Orbiter Reliability Block Diagram

T/D III

and there is a high probability of losing the battery due to a cell short. This would result in loss of data during eclipse. The last part of the mission which occurs in full sunlight would not be affected.

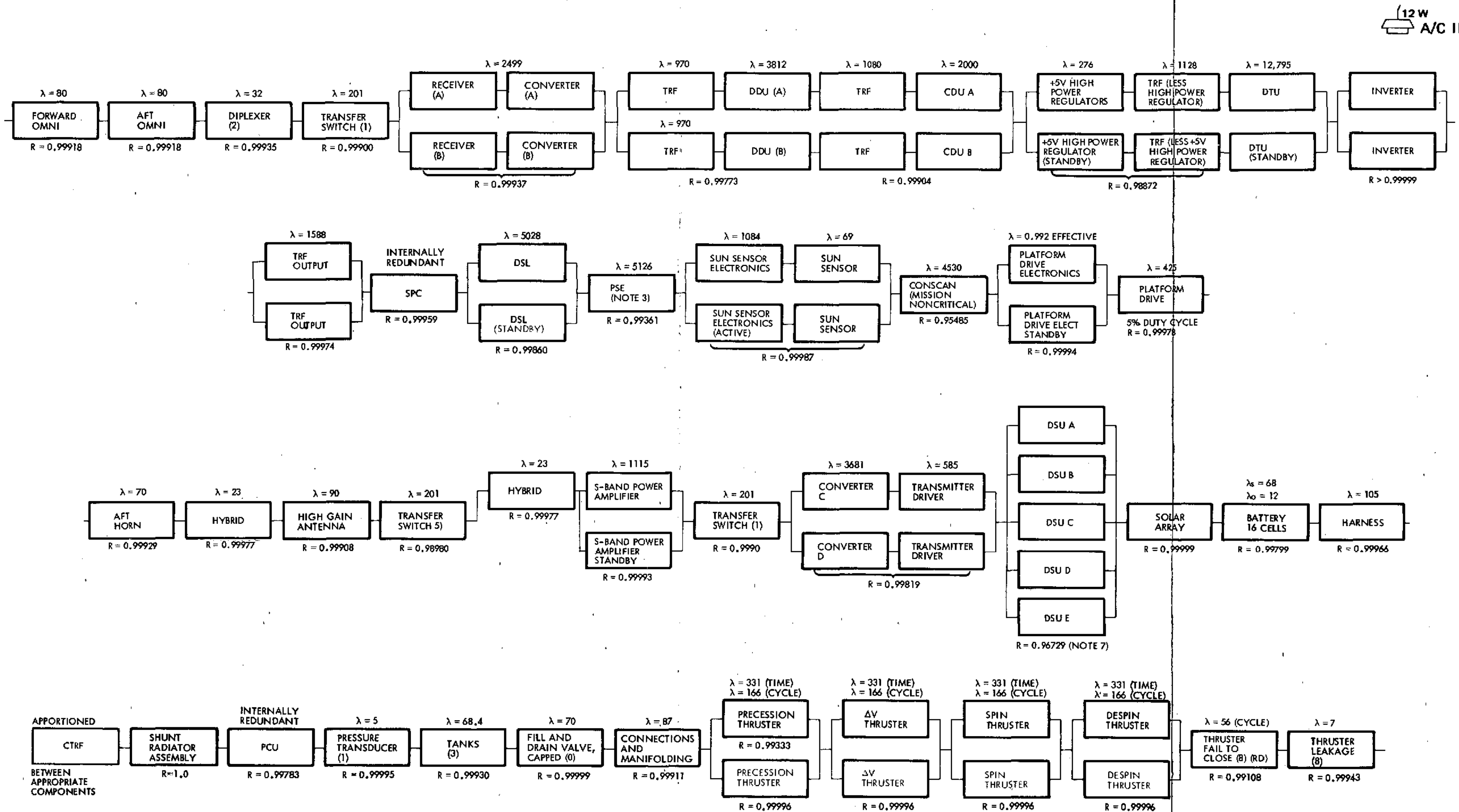
- Three data storage units (DSU) were added to the data handling subsystem. Two of the three units are required for the first 40 days; only one of three is required for the remainder of the mission.
- The communication subsystem replaces the solid-state transmitter amplifier with TWTA's. In addition, the medium-gain antenna is replaced by high gain antennas with fanbeam and fanscan. A conscan processor has been added for attitude determination. The processor is not mission critical since the attitude determination could be performed using ground backup operations, if necessary.

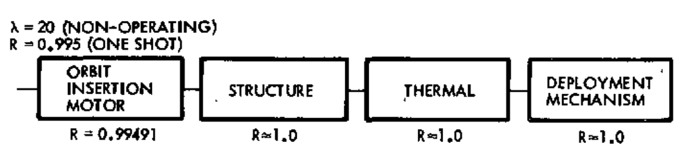
6.3.4.2 Atlas/Centaur Configuration 12W A/C III

The reliability block diagram for the preferred earth-pointing orbiter configuration is shown in Figure 6-60. The Atlas/Centaur orbiter reliability model is the same as for the probe bus (reference Section 6.3.3.3) except for the following changes:

- The electrical power subsystem has a different solar array design which has a 0.9999 probability of delivering 97 percent of the available array power (95 percent of available power is required). A nickel-cadmium battery replaces the silver-zinc type. During the first part of the Venus orbit mission the battery experiences a 30 percent depth of discharge. During this period there is very low probability of losing the battery due to a cell short. After 165 days of the Venus orbit mission and for a period of 25 days a very high depth of discharge occurs (80 percent) and there is a high probability of losing the battery due to a cell short. This would result in loss of data during the eclipse. The last part of the mission which occurs in full sunlight would not be affected. The transmitters and receivers have their own converter, eliminating output requirements from the CTRF, but additional CTRF output are required for the DSU's and conscan.
- The attitude control subsystem has an added platform for the radar altimeter which requires a gimbal drive and redundant gimbal drive electronics which are activated periodically during Venus orbit.
- The data handling subsystem has added five data DSU's of which four are required for the first 40 days and three are required for the remainder of the mission.
- The communication subsystem has dedicated converters for the transmitter drivers (one active, one standby), and for the receivers (both active). Also, a high-gain earth-pointing antenna has been added.

12 W
A/C III





ORBITER RELIABILITY = 0.908

ASSUMPTIONS:

1. MISSION TIME: 425 DAYS
2. TRANSIT TIME: 212 DAYS
3. PSE ONLY REQUIRED FOR 50 HOURS (ACTIVE) PLUS 1 HOUR PER ORBIT
4. THE CONSCAN IS CONSIDERED MISSION NONCRITICAL
5. FAILURE RATE (λ) IN FAILURES PER 10^9 HOURS
6. DSU TRANSIENT 200 DAY STANDBY; 55 DAYS 4 OF 5 REQUIRED FOR 8 HOURS ACTIVE 16 HOURS SA STANDBY; 170 DAYS 3 OF 5 REQUIRED FOR 8 HOURS ACTIVE 16 HOURS STANDBY
7. PIONEER 10 AND 11 FAILURE RATE
8. STANDBY K FACTOR OF 10 PERCENT

Figure 6-60. Atlas/Centaur Orbiter Relibility Block Diagram